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(2)

FIRST UPDATE OF AIRCRAFT ICING HANDBOOK

To Users of the Aircraft Icing Handbook:

Enclosed is the first update to the Aircraft Icing Handbook (DOT/FAA/CT-88/8-1, 2, 3). An overview of what has been changed is given in "INTRODUCTION TO THE FIRST UPDATE (9/93)" in the enclosed material. Detailed information on the insertion of the new material is contained in the document control sheet.

Many of the changes are in response to comments of users of the handbook. These comments were greatly appreciated, and further comments on the updated handbook will be equally welcome. The mailing address for comments is given in the notice which appears at the front of each volume of the handbook.

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PREFACE

This document, FAA Technical Report No. DOT/FAA/CT-88/8, produced in three volumes, is the final report of a program conducted by the Gates Learjet Corporation of Wichita, Kansas, to develop an updated comprehensive multi-volume engineering handbook on aircraft icing. The work effort was directed towards producing a combined version of Federal Aviation Administration (FAA) Technical Reports Number ADS-4 (airframe icing) and RD-77-76 (engine icing), which would include reference material on ground and airborne icing facilities, simulation procedures, and analytical techniques and represent all types and classes of aircraft. The program was sponsored by the FAA Aircraft Icing Program, Flight Safety Research Branch, at the FAA Technical Center, Atlantic City International Airport, New Jersey. Mr. Ernest Schlatter, Research Meteorologist, was the Technical Monitor for the FAA Technical Center. Dr. James T. Riley and Mr. Charles O. Masters of the FAA's Aircraft Icing Program were instrumental in the completion of this document following the retirement of Mr. Schlatter.

Work was performed under the coordination of Mr. A M. Heinrich, Project Director. Technical support was provided by Mr. Richard Ross of Ross Aviation Associates, Sedgwick, Kansas; Dr. Glen Zumwalt of Wichita State University, Wichita, Kansas; Mr. John Provorse of Cedar Hill Industries, El Dorado, Kansas; and Dr. Viswa Padmanabhan of the Gates Learjet Corporation.

Additional subcontractor contribution to the handbook was provided by the following companies and academic institutions:

Aeromet, Inc., Tulsa, Oklahoma
Beech Aircraft Corporation, Wichita, Kansas
Boeing Vertol Company, Ridley Park, Pennsylvania
Douglas Aircraft Company, Long Beach, California
Ideal Research, Inc., Rockville, Maryland
Kohlman Aviation, Lawrence, Kansas
Ohio State University, Columbus, Ohio
Pratt and Whitney Aircraft Group, East Hartford, Connecticut
Rosemount Inc., Burnsville, Minnesota
Rohr Industries, Inc., Chula Vista, California
Sikorsky Aircraft Company, Stratford, Connecticut
Sverdrup Technology, Inc., Tullahoma, Tennessee
Wichita State University, Wichita, Kansas

An additional contractor contribution to the handbook was provided by Mr. James Thompson of Thompson Enterprises, Arlington, Virginia.

Appreciation is gratefully extended for the assistance and material provided by personnel of the B.F. Goodrich Company, Akron, Ohio; Fluidyne Engineering Corporation, Minneapolis, Minnesota; National Research Council (NRC), Ottawa, Canada; NASA Lewis Research Center,

Cleveland, Ohio; U.S. Army Cold Regions Research and Engineering Laboratory (CRREL), Hanover, New Hampshire; Mount Washington Observatory, Gorham, New Hampshire; Leigh Instrument Company, Ontario, Canada; and McKinley Climatic Laboratory, Eglin AFB, Florida. Technical review was provided by the FAA's Aircraft Icing National Resource Specialist in Washington, D.C.; personnel in the four FAA Aircraft Certification Directorates (Boston, Massachusetts; Ft. Worth, Texas; Kansas City, Missouri; and Seattle, Washington); personnel at the NASA Lewis Research Center, Cleveland, Ohio, the U.S. Naval Research Laboratory, Washington, D.C.; and members of the Society of Automotive Engineers Aircraft Icing Technology Subcommittee AC-9C.

The Handbook is organized into nine major Chapters as follows:

VOLUME I

VOLUME II

VOLUME III

INTRODUCTION TO THE FIRST UPDATE (9/93)

The first update of the Aircraft Icing Handbook does not change the organization of the handbook into volumes and chapters as delineated on page vii. However, new sections have been added to Chapter III - Ice Protection Methods. Section 1.0 no longer contains information on pneumatic impulse de-icing and has therefore been retitled:

SECTION 1.0 - CONVENTIONAL PNEUMATIC BOOT DE-ICING SYSTEMS.

Three new sections have been added:

SECTION 1A.0 - PNEUMATIC IMPULSE DE-ICING SYSTEMS

SECTION 4A.0 - ELECTRO-EXPULSIVE DE-ICING SYSTEMS

SECTION 4B.0 - EDDY CURRENT DE-ICING SYSTEMS

Some changes have been made to all the sections in this Chapter with the exception of Section 3. All of Chapter VIII, the bibliography, has been replaced. Changes have also been made to parts of Section 1 of Chapter I, Sections 1 and 2 of Chapter IV, Section 4 of Chapter V, and Section 1 of Chapter VI.

New sections, and sections that have changes on many of their pages, are indicated with the footer "Update 9/93" on every page. In cases where only a few pages of a section have been altered, the footer "Update 9/93" appears on only those pages.

SYMBOLS AND ABBREVIATIONS (CONTINUED)

Symbol	Description
α	Angle of attack (degrees)
β	Local impingement efficiency at a location on an airfoil or body, dimensionless
δ	Droplet diameter (μm)
μ	Viscosity (slugs/ft.-sec.)
ρ_a	Density of air (slugs/ft^3)
ρ_i	Density of ice (g/cm^3)
ρ_w	Density of water (g/cm^3)
τ	Icing time (duration of encounter), minutes

Subscripts

a	Air
i	Ice
l	Local
L	Lower surface
med	For the median volume droplet diameter
max	Maximum
s	Static condition
U	Upper surface
w	Water.
∞	Denotes freestream conditions

Superscripts

(-)	Indicates impingement terms computed over a droplet spectrum
(.)	Derivative with respect to time

GLOSSARY

flux - Rate of flow per unit area.

impingement efficiency curve (β-curve) - A plot of the local impingement parameter β versus the surface distance parameter S for an airfoil or other two-dimensional object.

liquid water content (LWC) - The total mass of water contained in all the liquid cloud droplets within a unit volume of cloud. Units of LWC are usually grams of water per cubic meter of air (g/m^3).

median volumetric diameter (MVD) - The droplet diameter which divides the total water volume present in the droplet distribution in half; i.e., half the water volume will be in larger drops and half the volume in smaller drops. The value is obtained by actual drop size measurements.

micron (μm) - One millionth of a meter.

stagnation point - The point on a surface where the local free stream velocity is zero. It is also the point of maximum collection efficiency for a symmetric body at zero degrees angle of attack.

β-curve - See "impingement efficiency curve."

Supercooled water droplets in the atmosphere usually have diameters of less than 60 microns and experience Reynolds numbers small enough to permit their treatment as essentially spherical. (Although this is the universal computational practice, it has been argued that a droplet experiencing large accelerations in the vicinity of an ice accretion may assume a non-spherical shape which would alter its coefficient of drag and hence its trajectory (reference 2-3).)

Consider the trajectory of a single droplet approaching a body. The droplet trajectory equation is obtained by applying Newton's Second Law, F = ma, to the droplet. This equation can be expressed as

$$m\frac{d^2\vec{x}}{dt^2} = \vec{P} + \vec{M}_a + m\vec{g} + \vec{B} + \vec{D} \qquad (2\text{-}1)$$

where x is the position vector of the droplet, t is time (the acceleration a is of course equal to the second derivative of x with respect to time), P is the pressure gradient term, M_a is the apparent mass term, mg is the gravity force or "settling" term, B is the Bassett (unsteady) history force, and D is the drag force. The forces P and M_a are ordinarily neglected because the density of the particle (water droplet) is much greater that of the fluid (air) and the force mg can be neglected because of the very small mass of supercooled water droplets.

The force B accounts for the deviation of the flow pattern around the particle from that of steady state and represents the effect of the history of the motion on the instantaneous force (reference 2-4). It is essentially a correction to the drag term for an accelerating sphere. An accelerating sphere experiences a lower drag coefficient since it takes the flowfield some finite time to respond to the changing velocity and droplet Reynolds number. The term is significant if the particle density is of the same order as that of the fluid (which is not the case here), or if the particle experiences "large" accelerations. Droplets experience their largest accelerations when in the leading edge region of an airfoil, and the accelerations are larger yet if "glaze horns" are present. Norment (reference 2-5), using the work of Keim (reference 2-6) and Crowe (reference 2-7), has argued that for the icing problem the accelerations experienced by the droplets are not large enough for the Bassett term to be significant. Lozowski and Oleskiw (reference 2-8) included the Bassett term in the droplet trajectory equation used in their droplet trajectory and impingement code (Chapter IV, Section 2). They state that their results suggest that "in most cases ... the term may be ignored without severely affecting the accuracy of the calculations" (reference 2-8, p. 11). The Bassett force will be neglected in the rest of this discussion.

The drag term, D, can be expressed as

$$\vec{D} = \frac{1}{2}\rho C_D S\left|\vec{u} - \frac{d\vec{x}}{dt}\right|\left(\vec{u} - \frac{d\vec{x}}{dt}\right) \tag{2-2}$$

u is the local flowfield velocity vector, S is the cross sectional area of the sphere (or the projected frontal area of the sphere), and C_D is the drag coefficient. Note that the drag is evaluated using the velocity of the droplet with respect to the local airstream; this is sometimes called the "slip velocity."

All the terms on the right hand side of equation 2-1 other than D are now dropped and equation 2-2 is used to substitute for D; this yields

$$\frac{d^2\vec{x}}{dt^2} = \frac{3}{4}\frac{\rho_a C_D}{\delta\rho_w}\left|\vec{u} - \frac{d\vec{x}}{dt}\right|\left(\vec{u} - \frac{d\vec{x}}{dt}\right) \tag{2-3}$$

where the equation has oeen divided by the mass m of the droplet, δ is the droplet diameter, and

$$\rho_w = \text{droplet density},$$

$$\rho_a = \text{air density}.$$

A standard drag curve (figure 2-1) for a sphere has been established by bringing together experimental results from many sources (reference 2-9). Only a limited range of this curve need be fit for supercooled water droplets, since the relevant droplet Reynolds numbers rarely exceed 500. A number of different fits are available, some of which are discussed in reference 2-1.

2.2.1.2 Modified Droplet Inertia Parameter

Equation (2-3) will now be nondimensionalized in order to introduce the inertia parameter K and modified inertia parameter K_0 (both further discussed in Chapter IV, Section 2). Letting x and y be the components of the vector $\vec{x}$, define the nondimensional variables $x^* = x/c$, $y^* = y/c$, $t^* = t/(c/V_\infty)$, where c is a characteristic length, t is time, and V_∞ is the freestream airspeed. If the asterisks are suppressed after the equation is suitably rearranged, the nondimensional equation is

$$\frac{d^2\vec{x}}{dt^2} = \frac{1}{K}\frac{C_D Re_l}{24}\left(\vec{u} - \frac{d\vec{x}}{dt}\right) \tag{2-4}$$

Now $\overline{x}$ is the dimensionless droplet position vector, $\overline{u}$ is the dimensionless local flowfield velocity vector, t is nondimensional time, Re_l is the local relative droplet Reynolds number given by

$$Re_l = \frac{\rho_a\delta\left|\vec{u} - \frac{d\vec{x}}{dt}\right|}{\mu_a} \tag{2-5}$$

(μ_a is the viscosity of air) and K is the droplet inertia parameter given by

$$K = \frac{1}{18}\frac{\delta^2 V_\infty \rho_w}{c\mu_a} \tag{2-6}$$

It can be seen from equation (2-4) that the trajectory depends upon K and $C_D Re_l/24$. But $C_D Re_l/24$ can be shown (reference 2-2) to depend approximately upon Re, the free stream droplet Reynolds number which is given by

$$Re = \frac{\rho_a V_\infty \delta}{\mu_a} \tag{2-7}$$

Therefore the droplet trajectory depends approximately upon Re and K only.

Langmuir and Blodgett (reference 2-2) combined Re and K into a single parameter K_0, referred to as the modified inertia parameter, as follows:

$$K_0 = K\left(\frac{\lambda}{\lambda_s}\right) \tag{2-8}$$

The quantity in brackets, referred to as the range parameter, is the ratio of the trajectory distance of a droplet in still air, with an initial Reynolds number of Re and gravity neglected, divided by the *trajectory distance if the drag is assumed to obey Stokes law.* Using numerical methods, they obtained a graph giving the range parameter as a function of Re (figure 2-2).

Bragg (reference 2-10) has interpreted K_0 by rewriting Equation 2-4 as

$$\left[\frac{K}{C_D Re_l/24}\right]\frac{d^2\bar{x}}{dt^2} = \bar{u} - \frac{d\bar{x}}{dt} \tag{2-9}$$

If some suitable average of the term in brackets on the left can be found over the entire trajectory, the droplet path becomes a function of just this single variable. Under typical icing conditions K_0 can be interpreted as such an average. Bragg also derived the following expression:

$$K_0 = 18K\left[Re^{-2/3} - \frac{\sqrt{6}}{Re}\arctan\left(\frac{Re^{1/3}}{\sqrt{6}}\right)\right] \tag{2-10}$$

Equation 2-10 is shown in reference 2-1 to be within 1 percent of Langmuir's calculated values until Re approaches 1000 (much larger than the values for supercooled cloud droplets), where Langmuir's values diverge.

The approximate similarity parameter K_0 has been introduced here because of its wide use in icing calculations. As shall be seen, it greatly simplifies the presentation of droplet impingement data. K_0 will be further discussed in Chapter IV, Section 2, where ice scaling is addressed and where experimental and computational evidence will be presented in support of the use of K_0.

K_0 can be interpreted as relating the importance of droplet inertia to the importance of droplet drag forces. For small values of K_0, drag predominates and the droplet tends to follow the flow streamlines until very close to the body. If K_0 is small enough ($\approx .005$), the droplet acts approximately as a flow tracer. For large values of K_0, droplet inertia predominates and the droplet departs considerably from the flow streamlines as the body is approached. If K_0 is large enough (≈ 1.0), the droplet trajectory is approximately a straight line that intersects the body. Figure 2-3 shows trajectories for two droplets approaching an airfoil, one with a diameter of 5 μm and $K_0 = .011$ and the other with a diameter of 50 μm and $K_0 = .467$. The trajectories were computed with the computer code LEWICE, which is discussed in Chapter IV, Section 2. Four chord lengths in front of the airfoil, the two trajectories are coincident (not shown in figure); however, they diverge dramatically in the vicinity of the airfoil due to the large difference in K_0 between the two drops.

Bragg (reference 2-10) has derived another trajectory similarity parameter, $\bar{K}$, for which he has given a theoretical justification but which, nonetheless, has not as yet been widely adopted by other workers. K_0 and $\bar{K}$ are closely related and, differing by a constant factor if a simple drag law is used in deriving K_0. Bragg shows that $\bar{K}$ is given approximately by

$$\bar{K} = \frac{1}{18}\left(\frac{\rho_\infty^3 \delta^5 V_\infty}{c^3 \mu_a^2 \rho_a}\right)^{\frac{1}{3}} \qquad (2\text{-}11)$$

It follows, since K_0 approximates $\bar{K}$, that K_0 is approximately proportional to $\delta^{5/3}$, to $V_\infty{}^{1/3}$, and to $1/c$.

Figure 2-4 presents values assumed by K_0 under a range of MVDs and velocities that might be experienced by a general aviation aircraft in flight. The bottom panel is for a chord size (5.58 feet) representative of a full scale wing and the middle panel is for a chord size (3.1 feet) representative of a full scale horizontal stabilizer (both for a general aviation aircraft) while the top panel is for a chord size (6 inches) representative of an airfoil model. (Much research has been done with models of approximately this size, although larger models are preferred in tunnels which can accommodate them.)

Comparison among the three panels shows that K_0 is a strong function of chord size. In fact, the largest value of K_0 for a full scale wing is approximately equal to the smallest value of K_0 for the model. Examination of any one of the three panels also shows that K_0 varies strongly with MVD but much more weakly with aircraft velocity. All these observations are in accordance with equation 2-11. The reader may find it useful to refer back to this figure when studying the graphs presented later in which the impingement parameters E and β_{max} (defined in the next section) are presented as functions of K_0.

Figure 2-5 is constructed in the same manner, but using typical maximum droplet diameters rather than MVDs. It is interesting to note that the contrast among the three panels is now more pronounced due to the strong sensitivity of K_0 to droplet diameter. Now the largest values of K_0 even

for a full scale horizontal stabilizer are substantially smaller than the smallest values of K_0 for the model. This figure may be useful in interpreting the later graphs in which the impingement parameters S_U and S_L (defined in the next section) are presented as functions of K_0.

Example 2-1

An example of the calculation of K_0 for an airfoil is now presented.

Airfoil:	$c = 3.1$ foot chord - NACA 0012
Flight Speed:	$V_\infty = 200$ kt (230.16 mph)
Altitude:	$h = 10{,}000$ ft (pressure altitude)
Ambient Temperature:	$T = -4\ ^\circ F = 455.7\ ^\circ R$
Droplet Size:	$\delta = 20$ microns

First find the air density and viscosity.

From the given pressure altitude, P = 1455.6 psf (10.109 psi). Solve for the air density using:

$$\rho_a = \frac{P}{RT}$$

$$\rho_a = \frac{1455.6}{(1716)(455.7)} = .001862\,\frac{\text{slug}}{\text{ft}^3}$$

For viscosity, one can use the approximate relation:

$$\mu_a = 7.136 \times 10^{-10}\,T$$

$$\mu_a = 7.136 \times 10^{-10}(455.7) = .3252 \times 10^{-6}\,\frac{\text{slug}}{\text{ft-s}}$$

Now calculate Re and K.

In these units Re is given by:

$$Re = 5.537 \times 10^{-6}\,\frac{\rho_a V_\infty \delta}{\mu_a}$$

$$Re = 5.537 \times 10^{-6}\,\frac{.001862(200)20}{.3252 \times 10^{-6}} = 126.8$$

In these units K is given by:

$$K = 1.985 \times 10^{-12}\,\frac{\rho_w \delta^2 V_\infty}{c\mu_a}$$

$$K = 1.985 \times 10^{-12}\left[\frac{(1)(20)^2 200}{(3.1)(.3252 \times 10^{-6})}\right] = .155$$

If Langmuir and Blodgett's graphical method is used, the problem is completed by using figure 2-2, which shows that for Re = 126.8 the range parameter is approximately equal to .32. Then

$$K_0 = K(\frac{\lambda}{\lambda_s}) = (.155)(.32) = .050$$

Alternately, if Bragg's result is used, calculate K_0 using Equation 2-10:

$$K_0 = 18(.155)K\left[(126.8)^{-2/3} - \frac{\sqrt{6}}{126.8}\arctan\left(\frac{(126.8)^{1/3}}{\sqrt{6}}\right)\right] = .050$$

Summarizing this procedure for the usual case where the aircraft geometry, flight speed, pressure altitude, droplet size, and temperature are known:

1) From a standard atmospheric table obtain P from the pressure altitude, h.

2) Calculate the density (T in °F):

$$\rho_a = \frac{P}{(1716)(459.67+T)}\frac{\text{slugs}}{\text{ft}^3} \qquad (2\text{-}12)$$

3) Calculate the viscosity (T in °F):

$$\mu_a = 7.136\times10^{-10}(T+459.67)\frac{\text{slugs}}{\text{ft-sec}} \qquad (2\text{-}13)$$

4) Solve for the droplet freestream Reynolds number:

$$Re = 5.537\times10^{-6}\frac{\rho_a V_\infty d}{\mu_a} \qquad (2\text{-}14)$$

5) Solve for the droplet inertia parameter:

$$K = 1.985\times10^{-12}\frac{\rho_w \delta^2 V_\infty}{c\mu_a} \qquad (2\text{-}15)$$

6) Use Re and K to calculate the modified inertia parameter using either equation 2-8 and figure 2-2 or else using equation 2-10.

2.2.1.3 Droplet Impingement Parameters

Several impingement parameters can be defined to characterize the impingement properties of an airfoil or cylinder with respect to the cloud it encounters.

Figure 2-6 illustrates the definition of the impingement parameters S_U, S_L, ΔY_0, h, and E for an airfoil in a supercooled cloud. Let S denote arc length measured along the airfoil surface. It is conventional to take S = 0 at the leading edge, and that is done here (although the reader should note that it is sometimes convenient to take S = 0 at the stagnation point instead). S is defined to be positive on the upper surface and negative on the lower surface. S_U and S_L are defined to be the upper and lower limits of droplet impingement on the airfoil and are determined by the upper and lower tangent droplet trajectories. Define a Y-axis that is perpendicular to the freestream velocity and far enough in front of the airfoil (at least several chords) so that the flow is essentially undisturbed by the presence of the airfoil; then the droplet trajectories can be taken initially to be parallel to one another and to the freestream flow lines. The droplet trajectory which strikes the airfoil at its leading edge intersects the Y-axis at a point which is taken to be Y = 0. The upper tangent trajectory intersects the Y-axis at a point Y_U and the lower tangent droplet trajectory intersects it at a point Y_L. Let $\Delta Y_0 = Y_U - Y_L$; refer to this as the "freestream impingement width." Let h be the projected frontal height of the airfoil; note that this is a function of angle of attack. The total impingement (or collection) efficiency E is defined as the ratio of the freestream impingement width ΔY_0 to the projected frontal height h, i.e.,

$$E = \frac{\Delta Y_0}{h} \tag{2-16}$$

E is the proportion of liquid mass crossing the Y-axis within the frontal projection of the airfoil and ultimately striking the airfoil.

In equation (2-16), E is a dimensionless quantity, but ΔY_0 and h are not. However, it is customary to nondimensionalize the latter two quantities by dividing them by the chord length c. A different notation is not ordinarily introduced for nondimensional ΔY_0 and h; in instances where the meaning may not be clear from the context, it is explicitly noted if the dimensional or nondimensional quantity is meant. Tables and graphs are available giving nondimensional ΔY_0 and h as functions of K_0 and angle of attack α for some airfoils. Nondimensional ΔY_0 can be interpreted as follows: consider a segment of the Y-axis of length equal to one chord and centered at the projected position of the airfoil leading edge. ΔY_0 is the proportion of liquid mass crossing the Y-axis within the segment which ultimately strikes the airfoil.

Figure 2-7 illustrates the definition of the local impingement (or collection) efficiency ß at an arbitrary point P on the airfoil. Let P lie between the points of impact on the airfoil surface of two droplet trajectories. The mass of water droplets between the two trajectories a distance δY_0 apart in the free stream (at the Y-axis) is distributed over a length δS on the airfoil surface. Letting

δS approach 0 in such a way that P always falls between the impact points of the two trajectories, the local impingement efficiency β at P is defined in the limit by the derivative

$$\beta = \frac{dY_0}{dS} \tag{2-17}$$

The maximum value assumed by β anywhere on the airfoil surface is denoted by β_{max}. Note also that

$$\Delta Y_0 = \int_{s_L}^{s_U} \beta\, ds \tag{2-18}$$

The impingement efficiency curve or β-curve is a plot of β on the vertical axis versus S on the horizontal axis. This is illustrated in figure 2-8. The β-curve can be calculated numerically as follows: First, find the upper and lower tangent trajectories. These are ordinarily approximated numerically by finding upper and lower trajectories which pass within a small prescribed distance ϵ of the airfoil without actually striking it. Second, calculate a set of trajectories between the upper and lower trajectories (figure 2-9). There is a Y value and associated S value for each trajectory. Third, fit a Y vs. S curve to the points (S, Y), as shown in figure 2-10. Fourth, approximate the derivatives to the Y vs. S curve at a set of points; these derivatives are the βs. Fifth, fit a β-curve to the points (S, β). Some researchers omit step three and simply approximate β_i for (S_i, Y_i) by the ratio $(Y_{i+1} - Y_i)/(S_{i+1} - S_i)$, and then fit the β-curve to the points (S_i, β_i).

As noted, equations 2-16 and 2-17 are for the two-dimensional planar case. The local impingement efficiency, β, can be calculated for the three-dimensional case by considering a three-dimensional tube of water droplets starting at infinity with some area, A, perpendicular to the freestream, and impinging on a body over some surface area, A_s. Then, the local impingement efficiency, β, is the limit, as A_s approaches zero, of A divided by A_s:

$$\beta = \lim_{A_s \to 0} \frac{A}{A_s} \tag{2-19}$$

Discussions of three-dimensional impingement calculations can be found in references 2-11 and 2-5.

Example 2-2

This example illustrates the estimation of the impingement parameters E, β_{max}, h, S_U and S_L using graphical data (reference 2-12). The graphical data is all presented with K_0 as the independent variable. Much data is available in this form.

The conditions of Example 2-1 for a NACA 0012 airfoil are assumed: thus $K_0 = 0.05$. It also is assumed for simplicity that the angle of attack, α, is 0 degrees. From figure 2-11, E, the total impingement efficiency, is estimated to be 0.23 for these conditions. So about 23 percent of the water

in the projected frontal area of the airfoil, with height h = .120c (found using figure 2-12), impinges on the airfoil. The maximum impingement efficiency for K_0 = 0.05 and α = 0 degrees is estimated from figure 2-13 to be β_{max} = 0.68. At α = 0 degrees, the upper and lower limits of impingement are identical. From figure 2-14 at K_0 = 0.05, $S_U = S_L \approx .04$. Therefore, water droplets will impinge on the airfoil leading edge only back approximately 4 percent of chord. (Note that this example assumes a "monodispersed" cloud, that is, a cloud in which all the droplets are of the same size. More realistic approaches are discussed in the following section.)

2.2.1.4 Droplet Size Distribution Effects

The discussion thus far has proceeded as though clouds consisted of droplets of a single size ("monodispersed" clouds). All actual clouds, whether in the atmosphere or the wind tunnel, possess a spectrum of droplet sizes. This is taken into account in the definition of β and E by integrating over the droplet spectrum. In calculations with experimental data, this leads to taking averages weighted by volume, with the droplet spectrum represented by a histogram. Terms computed over the droplet spectrum are sometimes indicated by writing a bar above them.

$$\bar{\beta}(S) = \int_{\delta_{min}}^{\delta_{max}} \beta(\delta,S)\frac{dv}{d\delta}d\delta \qquad (2\text{-}20)$$

Here $\bar{\beta}$ is *called the droplet spectrum local impingement* (or collection) efficiency at the surface position specified by S. The phase "droplet spectrum" is ordinarily suppressed, since this is the meaning carried by the bar; some authors use the term "overall" rather than "droplet spectrum." The integral limits are the minimum and maximum droplet diameter in the cloud. In general, a droplet size distribution is described by v, the cumulative volume of water in the cloud as a function of droplet diameter, δ. In equation 2-20, the derivative of this curve, $dv/d\delta$, appears. It is a function of the droplet size, δ, and, of course, the assumed cloud droplet distribution. Usually β and $dv/d\delta$ are not known as continuous functions of δ and equation 2-20 is then represented as a summation

$$\bar{\beta}(S) = \sum_{i=1}^{N} \beta(\delta_i,S)\Delta v_i \qquad (2\text{-}21)$$

Equation 2-21 is summed over N discrete droplet sizes representing the midpoints of N droplet size bins. (For example, δ_i = 6.5 μm for a bin for droplets with diameters from 5 to 8 μm.)

The droplet spectrum (or overall) impingement (or collection) efficiency $\bar{E}$ for an airfoil or body is defined in a similar way for a droplet size distribution:

$$\bar{E} = \frac{1}{h}\int_{\delta_{min}}^{\delta_{max}} \Delta Y_0(\delta)\frac{dv}{d\delta}d\delta \qquad (2\text{-}21)$$

Here $\Delta Y_0(\delta)$ is the initial Y difference for the tangent trajectories for a droplet of diameter δ. As in the case of β, one usually knows $\Delta Y_0(\delta)$ for a discrete number of droplet sizes. Equation 2-22 can therefore be written as the sum

$$\bar{E} = \frac{1}{h}\sum_{i=1}^{N} \Delta Y_0(d_i)\,\Delta v_i \tag{2-23}$$

Note that a droplet spectrum (or overall) $\overline{\Delta Y_0}$ may also be defined as

$$\overline{\Delta Y_0} = \bar{E}h \tag{2-24}$$

The limits of impingement depend not on the entire droplet spectrum but only on the largest droplets present in the spectrum. Let δ_{max} denote the largest drop diameter present in the spectrum (or the midpoint of the bin containing the largest droplets), and let $K_{0,max}$ denote the modified inertia parameter calculated using δ_{max}. The maximum limits of impingement may be found from plots of S_U and S_L as a function of $K_{0,max}$ and angle of attack α.

Example 2-3

This example is a repetition of Example 2-2 except that this time the impingement parameters will be found using the entire droplet spectrum. It is assumed that the droplet median volume diameter (MVD) is 20 μm (the droplet size used in Example 2-2) and the cloud droplet spectrum can be represented by a Langmuir D distribution. The droplet sizes representing the seven size bins in the distribution are calculated using table 1-1 (discussed in Section 1.2.6). Table 2-1 shows the droplet sizes δ, the proportion Δv of total droplet volume associated with each δ, and also the values of Re, K, and K_0 for each δ. Using these values of K_0 and figure 2-11, a value $E(\delta)$ is associated with each δ, as shown in the third column of table 2-2. Note that equation 2-23 can also be written as

$$\bar{E} = \sum_{i=1}^{N} E_i(d_i)\,\Delta v_i \tag{2-25}$$

Thus $\bar{E}$ is calculated as an average value of the $E(\delta)$ weighted by volume using the Δv's. Table 2-2 shows that a value $\bar{E} = 0.24$ is obtained, little different from the value $E = 0.23$ for the MVD. This is well within the accuracy of reading numbers from the figure.

Considering this droplet size distribution and using equation 2-21, $\bar{\beta}$ can be calculated for a surface length location of $S = 0$ (stagnation point). For the special case of a symmetric airfoil at zero degrees angle of attack, where $\bar{\beta}_{max}$ occurs at $S = 0$ for all K_0, equation 2-21 can be used directly to determine $\bar{\beta}_{max}$. In table 2-2 the calculation of $\bar{\beta}$ at $S = 0$ is summarized in the last two columns, obtained from figure 2-11. Here again the value of $\bar{\beta} = 0.65$ is close to the value for the MVD droplet size, where $\beta = 0.68$.

The maximum limits can be found from $K_{0,max}$ and figure 2-14. For the 44.4 micron droplet size, $K_{0,max} \approx 0.176$ and from the figure $S_U = S_L \approx 0.11$, or 11 percent of the chord length.

Estimates of the size of a pneumatic ice protection boot have been made by using an MVD of 20 microns and twice that diameter (40 microns) to determine the maximum extent of significant droplet catch. This comes to about 10% of the airfoil chord on the critical upper surface. Ten percent coverage of the upper surface is consistent with statistical measurements of upper surface icing made in the USSR (reference 2-13).

Comparison of Example 2-2 and 2-3 suggests that, except for the limits of impingement, impingement parameters calculated using the MVD may give a reasonably good approximation to those calculated over an entire droplet distribution. This property of the MVD supplies the main justification for its wide use as the "representative" droplet size for a supercooled cloud in the study of aircraft icing. The error introduced in impingement calculations by its use rather than use of the full droplet spectrum is discussed in reference 2-14 and 2-15.

2.2.1.5 Approximate Two-dimensional Icing Formulas

Several approximate two-dimensional icing formulas are presented in this section. In these formulas, the symbols β, E and ΔY_0 for calculations with the MVD are used. If the corresponding quantities for the droplet spectrum are available, then bars are simply put over these quantities.

The formulas are approximate primarily for two reasons. First, freestream and ambient quantities are used throughout. Second, impingement parameters are for a clean body. As ice accretes, the shape of a body changes and with it the flow field and impingement parameters. For example, if the conditions were glaze and the duration long, a large glaze ice shape would actually accrete and some of the formulas here would give poor approximations. The formulas are most reliable for rime conditions or short durations.

Let $\dot{m}$ denote the water impingement rate per unit span in lbm/min-ft. span; then $\dot{m}$ is given by

$$\dot{m} = 6.322\times10^{-3}\, V_{\infty}(LWC)cEh \qquad (2\text{-}26)$$

where V_{∞} is the freestream velocity in knots, LWC is the liquid water content in g/m^3, c is the chord length in feet, E is the total collection efficiency (dimensionless), and h is the dimensionless projected height of the body. E and h would be found from a table or graph. Note that although E is a two-dimensional quantity, it may be used in a strip-theory type approach across the wing span as long as the sweep and induced angles of attack are taken into account.

Multiplying $\dot{m}$ by the duration τ of an icing encounter in minutes yields the mass m of water impingement per unit span, lbm/ft. span, for the encounter:

$$m = \dot{m}\tau \qquad (2\text{-}27)$$

A useful dimensionless term called the accumulation parameter, A_C, is introduced here. Its primary use is in the area of scaling, and it is more fully discussed in Chapter IV, Section 2. It is given by

$$A_c = \frac{LWC(V_\infty)\tau}{\rho_{ice} c} \tag{2-28}$$

where ρ_{ice} is the density of ice. Using the units of the variables as defined in the "Symbols and Abbreviations" and Example 2-4, A_C can be calculated using the following equation (reference 2-1):

$$A_c = 1.013x10^{-4} \frac{LWC(V_\infty)\tau}{\rho_{ice} c} \tag{2-29}$$

If the value of β is known at a point on the surface, the local ice thickness in chords may be approximated by

$$l = A_c \beta \tag{2-30}$$

This equation assumes that the ice growth is normal to the surface, so it is most accurate in the stagnation region and for blunt bodies. The maximum ice thickness in chords is given by

$$l_{max} = A_c \beta_{max} \tag{2-31}$$

For a cylinder or symmetric airfoil at 0° angle of attack, $\beta_{max} = \beta_0$, the impingement efficiency at the stagnation point; this may be available from a table or graph. Even for non-symmetric airfoils at 0° angle of attack, $\beta_{max} \approx \beta_0$.

The cross sectional area of an ice accretion can be approximated by

$$A = A_c E h = A_c \Delta Y_0 \tag{2-32}$$

where A is in units of chord length squared. Thus the area of the ice cross-section in chord lengths squared equals the area of a rectangle of length ΔY_0 and width A_C. Also note that $A_C E$ equals the area of the rectangle divided by the projected height of the airfoil, non-dimensionalized by airfoil chord (reference 2-1, p. 59).

Example 2-4

The mass of ice accretion on the NACA 0012 section will be calculated. Using the same flight conditions as Example 2-1, and the droplet size distribution and value from Example 2-3:

Airfoil:	$c = 3.1$ foot (37.2 in) chord NACA 0012
Flight Speed:	$V = 200$ knot
Airfoil Projected Height:	$h = .12$ at $\alpha = 0$ deg.
Liquid Water Content:	$LWC = 0.4\ g/m^3$
Collection Efficiency:	$E = .24$
Maximum Impingement Efficiency:	$\beta_{max} = .65$
Icing Time:	$\tau = 5$ minutes

Using equation 2-26 the mass of impinging water per unit span per unit time is given by:

$$\dot{m} = 6.322 \times 10^{-3}(200)(0.4)(3.1)(.24)(.12) = 0.045 \text{lbm/min-ft. span}$$

Then for a five minute icing encounter (equation 2-27)

$$m = .045(5) = 0.226 \text{lbm/ft. span}$$

Calculating the accumulation parameter from equation 2-29 gives:

$$A_c = 1.013 \times 10^{-4} \frac{(.4)(200)(5)}{.8(3.1)} = 0.0163$$

Note that the density of the ice is assumed to be 0.8 g/cm^3, implying a rime accretion.
The maximum ice thickness is approximated from equation 2-31 as

$$l_{max} = 0.0163(.65) = 0.0106c, \ c = \text{airfoil chord}$$

Thus the maximum ice growth is approximately 1.1 percent of the airfoil chord length, or about (.0106)(37.2) = .39 inches.
The cross sectional area of the ice in square chords, from equation 2-32, is

$$A = (.0163)(.24)(.12) = (.0163)(.0288) = 4.7 \times 10^{-4} c^2$$

The area in feet is given by

$$A = 4.7 \times 10^{-4}(3.1)^2 = 4.5 \times 10^{-3} ft^2$$

which is equal to about 0.65 in^2.

2.2.1.6 Compressibility Effects on Droplet Impingement

The droplet impingement characteristics of a body in a flow at high subsonic Mach number may differ from those in a flow at low Mach number. Brun, Serafini and Gallagher (reference 2-16) performed numerical calculations of the droplet impingement on a cylinder at M = 0.4 and compared these results to incompressible data of reference 2-17. The result of the calculations, performed over a K_0 range from 0.2 to .66, was a reduction in E, but never by more than 3 percent. The compressible and incompressible curves, shown in the lower right area of figure 2-15, are barely distinguishable from one another.

More recent data on a NACA 0012 airfoil at $\alpha = 0$ degrees and Mach numbers from 0 to 0.8 are also shown in figure 2-15. These data are from references 2-12 and 2-18, with most of the compressible data coming from the latter reference, where a compressible flowfield and a compressible form of the trajectory equation are employed. Figure 2-15 indicates that compressibility effects are greater for lower values of K_0; this presumably is due to the greater sensitivity of the droplets to flowfield changes at these lower K_0's. The effect of compressibility is to reduce E and also S_U and S_L (not shown). These limited studies suggest that design using the incompressible droplet impingement data may be conservative.

2.2.1.7 Droplet Impingement Data

In this section, a selection of droplet impingement data, both theoretical and experimental, will be presented. Refer to Sections 2.2.1.1 - 2.2.1.4 for definitions and uses of the data. Figure 2-16 gives the projected height of several airfoils as a function of angle of attack. Also note that table 2-3 provides the characteristic length used to calculate K_0 for the various bodies. This table is important since the characteristic length used in the calculation of K_0 is a matter of convention and the conventional choice is not always obvious. For example, for a cylinder the radius is used, whereas one might have expected the diameter, by analogy with the use of the chord for an airfoil.

Experimental Data

In the 1950's, the NACA carried out an extensive experimental program to provide impingement information for airfoils and other geometries. Their method required a wind tunnel with spray system and consisted of seven steps:

1. Put dye in the spray system.
2. Put strips of blotter paper at strategic locations on an airfoil, aircraft component, or other geometric object.
3. Expose the object to the spray for a fixed time interval.
4. Remove the strips and place each in a separate container of water.
5. Wait until approximately all the dye in the paper has dissolved in the water. (This could take weeks.)

-4°, 0°, 4°, 8°) and values of E, β_{max}, Y_0, S_U and S_L for several values of α and K_0 (e.g., α = -4°, 0°, 4°, 8°, and K_0 = .01, .05, .1, .5, 1.0). These tables can be used to approximate impingement parameters for arbitrary values of α and K_0 through interpolation, to compare impingement properties of two or more airfoils, and also to search for trends or patterns in the impingement parameters as a function of airfoil characteristics such as thickness, camber, and radius of curvature.

Figures 2-48 to 2-59 present a selection of results from reference 2-33: computed values of E, β_{max}, S_U and S_L are shown as a function of K_0 for three airfoil angles of attack. These figures are constructed using the tables discussed in the previous paragraph. Six airfoils from the thirty in the study were selected for the figures (figure 2-47). The NACA 0012 airfoil has been widely used in aircraft icing research during the 1980's. A number of airplanes have airfoils from the NACA 23 series, to which the NACA 23012 belongs. The NACA 63-415 is a "classic" NACA laminar flow airfoil. The NACA 64-109 has been used on the empennage of several general aviation airplanes. The LS(1)0417 and MS(1)0313 are perhaps the most widely used of the "modern" laminar flow airfoils. In summary, the NACA 64-109 and NACA 0012 would be used for the empennage but not for a wing; the other four airfoils would be used for a wing.

Use of figures 2-48 through 2-53 will be illustrated by discussing their use for MVD of 20 μm, an airspeed between 100 and 220 knots, and a chord size of 5.58 feet (representative of an airfoil section for a wing of a full scale general aviation aircraft). As seen from figure 2-4, a value of $K_{0,MVD} \approx .02$ roughly corresponds to these conditions. Thus in examining figures 2-48 (α = 0°), 2-49 (α = 4°), and 2-50 (α = 8°) for E, one focuses on the lower left hand corner of the graphs. There is very little variation in this region among the airfoils and E is approximately equal to .10 or less for all of them in this region. Note that there is relatively little variation among the airfoils in these figures for <u>all</u> values of K_0, the curves for the NACA 64-109 being the ones that most stand out. The NACA 64-109 is the thinnest of the airfoils with a thickness of 9 percent of chord, all the rest having a thickness of at least 12 percent of chord.

In examining figures 2-51 (α = 0°), 2-52 (α = 4°), and 2-53 (α = 8°) for β_{max}, one notes more variation among the airfoils, with the curves for the NACA 64-109 quite different from those for the others. For $K_{0,MVD} \approx .02$, β_{max} is in the vicinity of .40 for all airfoils except the NACA 64-109, where it is in the vicinity of .60. Note also that there is a tendency of β_{max} to decrease with increasing α. As noted in Chapter IV, Section 2, accurate calculation of β_{max} for such a value of K_0 is a computational challenge, particularly at higher angles of attack. Thus some of the variation among the airfoils in these figures is certainly numerical; how much is not known.

Use of figures 2-54 through 2-59 will be illustrated by discussing their use for a maximum droplet diameter of 45 μm (which approximately corresponds to the maximum diameter for a Langmuir D distribution with an MVD of 20 μm), along with an airspeed between 100 and 220 knots and a chord size of 5.58 feet, as before. As seen from figure 2-5, a value of $K_{0,max} \approx .1$ roughly corresponds to these conditions. Figure 2-54 indicates that all the airfoils have an S_L value of about .10 or less at these conditions at α = 0°. The figure shows relatively little variation among the airfoils.

Figures 2-55 (α = 4°) and 2-56 (α = 8°) show a dramatic upward shift in the curves as one would expect, since the impingement on the lower surface will greatly increase as the angle of attack is increased.

In examining figures 2-57 (α = 0°), 2-58 (α = 4°), and 2-59 (α = 8°) for S_U, one notes considerably more variation among the airfoils that was the case for S_L. This is perhaps to be expected, since as the angle of attack is increased, what little impingement occurs on the upper surface will presumably be quite sensitive to the shape of the airfoil. For $K_{0,Max} \approx .1$, S_U is in the vicinity of .07 for all airfoils at α = 0° (figure 2-57), in the vicinity of .01 at α = 4° (figure 2-58), and still in the vicinity of .01 (although somewhat smaller) at α = 8° (figure 2-59). Accurate calculation of S_U for such a value of K_0 is also a computational challenge, particularly at higher angles of attack, so some of the variation among the airfoils in these figures is also numerical.

Comparison of Impingement Properties of a Circular Cylinder and a NACA 0012

Using figures 2-43 and 2-44 for a circular cylinder along with figures 2-11 and 2-13 for the NACA 0012, it is possible to compare the impingement properties of a blunt or bluff body (the cylinder) with those of a streamlined body (the NACA 0012 airfoil). The diameter of the cylinder is taken to be equal to the chord of the airfoil. For illustrative purposes, a very small airfoil model will be assumed, with a chord size of 5 cm. As indicated in table 2-3, K_0 is computed using radius (rather than diameter) for a cylinder while it is computed using chord for an airfoil. Assume that conditions are such that $K_0 = .2$ for the cylinder; then $K_0 = .1$ for the airfoil.

According to figures 2-43 and 2-13, $\beta_{max} \approx .30$ for the cylinder and $\beta_{max} \approx .78$ for the airfoil. According to equation 2-31, it follows that, for a small rime accretion, the ice thickness at the stagnation point would be about two and a half times greater for the airfoil. According to figures 2-44 and 2-11, $E \approx .05$ for the cylinder and $E \approx .37$ for the airfoil. Recall that $E = \Delta Y_0/h$. For the cylinder, the projected frontal length h is equal to the diameter of the cylinder which, nondimensionalized by the diameter itself, is simply equal to 1; hence $\Delta Y_0 = (.05)(1) = .05$ (dimensionless). For the NACA 0012 airfoil at α = 0°, h is equal to the airfoil thickness which, nondimensionalized by the chord, is equal to .12; hence (.78)(.12) = .044 (dimensionless). Assume the accreted ice is directly proportional to the impinging mass (which should be approximately true for a rime accretion with no water loss due to splashing or shedding). Then, according to equation 2-26, the mass of accreted ice is directly proportional to $\Delta Y_0 = Eh$, and so more ice will accrete on the cylinder than on the airfoil. This apparent contradiction is resolved when it is realized that far less mass impinges on the cylinder than on the airfoil relative to their respective projected frontal areas.

Note also that impingement curves for cylinders have different general shapes than those for airfoils. An airfoil impingement curve is often narrow and peaked in the stagnation region and for most conditions has a region that is distinctly concave upward. A cylinder impingement curve is not peaked in the stagnation region and the entire curve is concave down or only slightly concave upward toward the limits of impingement.

This model is formulated computationally by dividing the airfoil surface into segments, and associating a control volume with each segment. The water entering a control volume has two sources: (1) water droplets impinging on the surface segment; (2) water "running back" from an adjacent control volume closer to the stagnation point. (This "run back" water consists of all water which entered the adjacent control volume but did not freeze.) An energy balance analysis is applied to each control volume to determine the freezing fraction n, the fraction of the incoming water which freezes for that control volume. If $n = 1$, then all incoming water freezes. If $n < 1$, then a fraction $1-n$ does not freeze. This water will in turn run back into the adjacent control volume further away from the stagnation point.

The mass and energy balance analyses for a given control volume will now be presented in some detail. The energy balance analysis was given its classic formulation by Messinger (reference 2-41), whose work drew on earlier work by Tribus (reference 2-42). The presentation and notation used here is based on reference 2-43.

The mass balance for a control volume on the surface can be formulated as follows (figure 2-73). Let $\dot{M}''_{Imp}$ and $\dot{M}''_{Evap}$ denote the mass flux per unit time due to the impinging water droplets and to evaporation, $\dot{M}''_{Run\ in}$ and $\dot{M}''_{Run\ out}$ denote mass flux per unit time into and out of the control volume due to liquid run back, and $\dot{M}''_{Ice}$ denote the mass of ice formed per unit area per unit time. Then the mass balance for the control volume is:

$$\dot{M}''_{Ice} = \dot{M}''_{Imp} + \dot{M}''_{Run\ in} - \dot{M}''_{Run\ out} - \dot{M}_{Evap} \qquad (2\text{-}33)$$

The term $\dot{M}''_{Imp}$ is given by:

$$\dot{M}''_{Imp} = V_\infty LWC\beta \qquad (2\text{-}34)$$

V_∞ is the freestream velocity. However, if the local velocity at the edge of the boundary layer is available, that velocity should be used rather than the freestream velocity. This procedure is followed, for example, in the ice accretion code LEWICE. ß is the local collection efficiency for the control volume.

It is convenient to define a term $\dot{M}''_{Incoming}$ by:

$$\dot{M}''_{Incoming} = \dot{M}''_{Imp} + \dot{M}''_{Run\ in} \qquad (2\text{-}35)$$

Then the freezing fraction n for the control volume is defined by:

$$n = \frac{\dot{M}''_{Ice}}{\dot{M}''_{Incoming}} \tag{2-36}$$

where $\dot{M}''_{Ice}$ is the incoming mass which freezes.

The energy balance for a control volume on the surface can be formulated as follows (figure 2-74). First, the main heat source terms (those that release heat into the control volume) are given.

Let $\dot{Q}''_{Freeze}$ denote the heat released by the freezing of the incoming water. Then

$$\dot{Q}''_{Freeze} = n\dot{M}''_{Incoming}L_f \tag{2-37}$$

where L_f is the heat of fusion.

Let $\dot{Q}''_{Aero\ Heat}$ denote the aerodynamic heating. Then

$$\dot{Q}''_{Aero\ Heat} = \frac{h_c r_c V_\infty^2}{2C_{p_{air}}} \tag{2-38}$$

where h_c is the local heat transfer coefficient, r_c is a recovery factor, and $C_{p_{air}}$ is the specific heat of air.

Let $\dot{Q}''_{Droplet\ K.\ E.}$ denote the kinetic energy of the incoming droplets. Then:

$$\dot{Q}''_{Droplet\ K.E.} = \frac{\dot{M}''_{Imp}V_\infty^2}{2} \tag{2-39}$$

Let $\dot{Q}''_{Ice\ Cool}$ denote the cooling of the ice to the surface temperature T_{Surf}. Then

$$\dot{Q}''_{Ice\ Cool} = n\dot{M}''_{Ice}(T_f - T_{Surf}) \tag{2-40}$$

where T_f is the ice/water equilibrium temperature (32 °F). Note that if $n < 1$, $T_{Surf} = T_f$ and so this term equals 0.

Define $\dot{Q}''_{Source}$ by:

$$\dot{Q}''_{Source} = \dot{Q}''_{Freeze} + \dot{Q}''_{Aero\ Heat} + \dot{Q}''_{Droplet\ K.E.} + \dot{Q}''_{Ice\ Cool} \tag{2-41}$$

Next, the main heat sink terms (those that remove heat from the control volume) are given.

Let $\dot{Q}''_{Conv}$ denote the convective cooling term. Then

$$\dot{Q}''_{Conv} = h_c(T_{Surf} - T_\infty) \tag{2-42}$$

where T_∞ is the freestream temperature. If the local temperature at the edge of the boundary layer is available, that temperature should be used in this term rather than the freestream temperature. This

is also done in LEWICE. Note: The term $\dot{Q}''_{Conv}$ is often defined by

$$\dot{Q}''_{Conv} = h_c(T_{Surf} - T_r)$$

where the "recovery temperature" T_r is given by

$$T_r = T_\infty + \frac{h_c r_c V_\infty^2}{2C_{P_{air}}}$$

In this formulation the term $\dot{Q}''_{Aero\ Heat}$ is omitted from equation (2-41). In subsequent calculations in this section, $\dot{Q}''_{Aero\ Heat}$ is retained and equation (2-42) is used to calculate $\dot{Q}''_{Conv}$.

Let $\dot{Q}''_{Drop\ Warm}$ denote the droplet warming term. Then

$$\dot{Q}''_{Drop\ Warm} = \dot{M}''_{Imp} C_w (T_{Surf} - T_\infty) \tag{2-43}$$

where C_w is the specific heat of water.

Let $\dot{Q}''_{Evap}$ denote the heat loss due to evaporation. There are a variety of formulations of this term. The approach used her is based on references 2-44 and 2-U1 and employs a form of the Reynolds analogy. $\dot{M}''_{Evap}$ is given by

$$\dot{M}''_{Evap} = g \Delta B \tag{2-44}$$

where g is the mass transfer coefficient times the air density and ΔB is the "evaporative driving potential" dependent on the vapor concentration difference between the surface and the edge of the boundary layer. These quantities are given by:

$$g = \frac{h_c}{C_{P_{air}}} \left(\frac{Pr}{Sc}\right)^{.667} \tag{2-45}$$

$$\Delta B = \frac{B_1}{B_2} \tag{2-46}$$

$$B_1 = \frac{P_{v,Surf}}{T_{Surf}} - \left(\frac{P_e}{P_\infty}\right)\frac{P_{v,\infty}}{T_e} \tag{2-47a}$$

$$B_2 = \frac{1}{0.622}\frac{P_e}{T_e} - \frac{P_{v,Surf}}{T_{Surf}} \tag{2-47b}$$

The Prandtl number Pr, Schmidt number Sc, and specific heat of air $C_{P,air}$ should be evaluated at the film temperature $(T_e + T_{surf})/2$. $P_{v,surf}$ is the vapor pressure at the surface and $P_{v,\infty}$ is the free stream vapor pressure. The equations assume that P_e and T_e, the free stream pressure and temperature at the edge of the boundary layer are available; if they are not, the corresponding freestream values are used. 0.622 is the ratio of the molecular weight of water to that of dry air. The heat loss due to evaporation is now given by:

$$\dot{Q}''_{Evap} = \dot{M}''_{Evap} L_v \tag{2-48}$$

L_v is the heat of vaporization.

If the freezing fraction is equal to 1 and the surface temperature $T_{Surf.}$ is to be computed, then $\dot{Q}''_{Evap}$ should be replaced by the heat loss due to sublimation, denoted by $\dot{Q}''_{Subl}$. This is given by

$$\dot{Q}''_{Subl} = \dot{M}''_{Subl} L_s \tag{2-49}$$

where $\dot{M}''_{Subl}$ denotes the mass flux due to sublimation per unit time and L_s denotes the heat of sublimation. In some programs, $\dot{M}''_{Subl}$ is computed using the same formulas as $\dot{M}''_{Evap}$.

Define $\dot{Q}''_{Sink}$ by:

$$\dot{Q}''_{Sink} = \dot{Q}''_{Conv} + \dot{Q}''_{Drop\ Warm} + \dot{Q}''_{Evap} \tag{2-50}$$

The energy balance equation is:

$$\dot{Q}''_{Source} + \dot{Q}''_{Sink} = 0 \tag{2-51}$$

The control volume freezing fraction is calculated as follows: Assume that the equilibrium temperature, T_{surf}, is T_f. With this assumption, all quantities in the energy balance except n can be evaluated. Now solve for n. If the calculation yields a value of n between 0 and 1 inclusive, the calculation is complete. If n is calculated to be larger than 1, assume that the excess over 1 is because T_{surf} is actually smaller than T_f. So set n equal to 1 in the energy balance equation, and solve it iteratively (since several quantities depend on T_{surf}) for T_{surf}. If n is calculated to be smaller than 0, sett n equal to 0 and solve iteratively for T_{surf}, which is now be larger than T_f.

A major source of uncertainty in calculating n using this equation arises from the uncertainty in the computation of the heat transfer coefficient h. If n is calculated in the stagnation region of a cylinder, it is common to use the heat transfer correlation for a smooth cylinder (given, for example, in reference 2-45). If n is to be calculated in the stagnation region of an airfoil, the same correlation is sometimes used with radius equal to the radius of curvature of the airfoil. As the ice accretes, the shape changes and the surface roughness also changes, perhaps increasing dramatically. This can have a profound effect on the heat transfer coefficient.

shown in tables 2-6a and 2-6b for increasing LWC, the reason being that increasing the droplet size has the effect of increasing the liquid water impacting the surface.

Finally, figures 2-78a and 2-78b illustrate the approximately linear decrease in n as the freestream airspeed V_∞ is increased from 70 m/s to 130 m/s for both conditions a and b. This is primarily the effect of aerodynamic heating. However, note that the dependence of n on V_∞ is more complicated than its dependence on the other variables. As V is increased, the collection efficiency increases and the contributions of the convective cooling and evaporation terms change, since both depend on the heat transfer coefficient h, and the calculation of h depends on V_∞. Note that n falls much more rapidly in figure 2-78b, which corresponds to the warmer temperature condition.

Tables 2-8a and 2-8b show the relative contributions to the energy balance of the main heat source and heat sink terms for conditions (a) and (b) as V_∞ increases. For the source terms, the relative contribution of the aerodynamic heating increases steadily in importance for both conditions (a) and (b) as V_∞ increases. Note, however, that it makes a much larger percentage contribution for condition (b), which has the smaller LWC of .1. As to the sink terms, comparison of the tables shows that the relative contribution of the droplet warming term is much smaller for the smaller LWC (condition (b)), with the convective and, especially, evaporative terms larger at its expense. For both tables, the relative contributions of the sink terms are nearly constant as V_∞ increases.

Figures 2-81 to 2-83 (reference 2-49) are based upon an analysis published in 1952 using the freezing fraction concept developed by Messinger (reference 2-41). The plots show an estimate of the freezing fraction for the stagnation line of a two-inch diameter cylinder in a cloud of 15 micron droplets at a 5,000 foot (1.5 km) altitude for LWCs of 0.2 g/m^3, 0.5 g/m^3, and 1.0 g/m^3. Freezing fraction lines are shown as a function of ambient temperature and true airspeed. These lines would undoubtedly shift if a different model were used. However, the general relationships and trends would remain the same, and it is to illustrate these that the figures are reproduced here. Note that the threshold temperature between types of ice ($n = 0$, $n = .66$ and $n = 1.0$) decreases both with increasing airspeed and with increasing liquid water content. The choice of $n = .66$ as a boundary between glaze and "intermediate" ice is arbitrary.

Criticisms of the Model

Criticism of this model has focused primarily on the runback assumption, not on the control volume energy balance analysis. Reference 2-50 did investigate the possibility that the control volume analysis should include an extra heat source term which would be proportional to the film thickness. However, it was concluded on the basis of an order of magnitude analysis that such a term would not have a major effect on the computation of the freezing fraction.

Much of the recent discussion of the need for revision of the model grew out of close-up movies (and stop action photographs) of the icing process made in the Icing Research Tunnel at the NASA Lewis Research Center (reference 2-51). The movies show surface phenomena at several positions on a symmetrical wooden airfoil immersed in a cloud in the tunnel. The airfoil had a 11.4 cm chord,

a 12 cm span, and a cylindrical leading edge of 1.9 cm radius. Tunnel runs were conducted for a range of airspeeds (50 to 320 km/hr), air temperatures (above freezing down to -25 °C), and cloud conditions. Material from the films was selected to produce a single film showing some of the most important and interesting results (reference 2-52).

The following discussion of the picture that emerged from these films is based on reference 2-5?. It is convenient to begin with the results at air temperatures above freezing, since these reveal the surface phenomena without the influence of freezing (figure 2-79a). Large surface drops (beads) are formed from the cloud droplets impacting the surface of the airfoil. When these drops are large enough, they start to move downstream. The lower the airspeed, the larger the drops before they start to move. As they move downstream, the larger drops apparently shed, since only smaller drops are observed on the surface downstream. The film sequences apparently do not show any flowing film of liquid; all liquid transport is through the movement of large drops.

When the above-freezing experiment was performed over a rough artificial ice surface, the same surface behavior was observed except that the surface drops grew larger before they moved.

For below-freezing temperatures and aircraft airspeeds (figure 2-79b), surface liquid transport is again confined to the movement of large drops (the biggest of which were observed at low airspeeds.) However, even the large drops move only in a region near the stagnation line and only during a short initial transient phase. The size of the region of large drop movement and the length of the initial transient phase both tend to increase with increasing sub-freezing temperatures and with decreasing airspeed. The stagnation region initially has a thin water film; away from the stagnation line, this film gives way to very large stationary drops on top of ice hills. The width of the thin-film region decreases with time, and increases somewhat with decreasing temperature. Refer to figures 2-80a and 2-80b for stop action photographs from the film.

The film of reference 2-53 has now been viewed by a large number of researchers. Two aspects of the picture sketched above are sometimes discussed. First, is it true that any large surface drops which are observed to move after the initial phase are in fact shed? Second, is it true that the thin water film in the region of the stagnation line does not contribute to any runback?

Reference 2-54 attempts to explain the existence of stationary surface drops (which this reference refers to as beads) in terms of contact angle and contact angle hysteresis. It is observed that the liquid beads were often surrounded by regions of otherwise dry surfaces. The strong temperature dependence of contact angle behavior indicates the potential importance of thermal gradients on the ice surface. Small variations in surface temperature could restrict the mobility of water and be the cause of the stable nature of surface water beads. A cold dry surface would impose a barrier to water flow away from a bead.

Based on the experimental observations of ice formation in the glaze ice regime, a Multi-Zone model, in which the accreting ice surface is divided into two or more discrete zones with varying surface roughness and water behavior, has been proposed by Hansman and his associates (references 2-55, 2-56, and 2-57). In the simplest version, the surface is divided into two zones, the smooth

suggest the following explanation. The rime iced lift curve of figure 2-104a is characteristic of trailing edge stall. This suggests that the mixed ice shape of figure 2-103 retained a small separation bubble which eventually "burst" for higher values of α. The rime ice of figure 2-104 may have tripped the boundary layer and prevented the formation of the leading edge separation bubble.

Figure 2-105 shows results for a 15° flap configuration. Note the ice deposition patterns on the lower surface.

The physics of ice accretion for the horizontal and vertical stabilizer is the same as for the wing. However, they are more efficient collectors of ice than the wing because they are of smaller chord and are ordinarily thinner and of smaller leading edge radius. The effect of the ice on lift and drag is similar to that for the wing, although the magnitude of the changes may differ, again because of geometrical differences from a typical wing. Ingelman-Sundberg and Trunov (reference 2-77) studied the effect of ice on a three-dimensional tail section with a NACA 64A-009 airfoil. Their results show that a thin airfoil with small leading edge radius is less severely affected by ice; this they ascribe to the already low $C_{L_{max}}$ of these sections. When the leading edge was modified so as to increase the $C_{L_{max}}$ of the clean section, the detrimental effect of ice on $C_{L_{max}}$ also increased.

During flight, the horizontal and vertical stabilizer will of course be at angles of attack different from the wing, will experience velocities differing from those at the wing, and may experience icing conditions differing in other ways from those at the wing as well. Of prime concern is the effect that ice on these surfaces has on aircraft stability, control and handling characteristics. This is discussed in Section 2.3.5, where special attention is given to ice contaminated tailplane stall (ICTS).

This section has emphasized empirical methods and experimental results. Theoretical methods are being developed and important improvements in these methods are being made. The current state-of-the-art in analytical methods will be presented in Chapter IV, Section 2.4.0.

2.3.2 Propellers

Propeller blade sections accrete ice and suffer a loss in aerodynamic performance in much the same way as wing and tail sections do. Propeller blades have a small chord and operate at a high effective velocity, thus greatly increasing the collection efficiency and mass of ice (relative to chord size) accreted. Due to the large centrifugal forces, ice shedding, particularly near the propeller blade tip, is a major consideration in any propeller blade icing analysis.

Little experimental work on propeller blade icing aerodynamics has been conducted in recent years. Propellers for aircraft which are certified for flight into icing conditions are usually protected, and apparently little work has been done to determine the effects of ice on propeller performance. However, some analytical work conducted specifically on the analysis of propellers in icing conditions is reported in reference 2-78. The method uses an airfoil icing correlation and a propeller aerodynamics code to predict icing effects on propeller performance. Miller (reference 2-79) used the Bragg, Gray and Fleming correlations with a computer code for comparison of propeller performance, checking against the Neel and Bright flight test data discussed in the following paragraph. This code produced realistic thrust and power coefficients, especially when the radial icing extent was known and input to the code.

Perhaps the best propeller icing data are the results of Neel and Bright (reference 2-80). Flight

tests were conducted with one propeller of a twin-engine aircraft allowed to collect ice and the other propeller kept ice-free. Ice thicknesses of up to one inch on the blade were measured. The spanwise extent of ice was from zero to 95 percent. Efficiency losses were less than 10 percent in most cases, with losses of 15 to 20 percent possible in some situations.

The analysis of ice effects on moving surfaces adds an additional dimension to the icing problem. Two components of motion can be important to this analysis. The first is rotational motion, which produces Mach number (and hence total temperature) variation across the blade span, and introduces centrifugal forces to shed ice. The second component of motion is present for a helicopter rotor but not for a propeller: the variation of blade angle during a revolution. This motion increases the chordwise extent of ice and may result in the operation of rotor blade sections into stall for a greater than normal portion of the rotor disk. Therefore, the calculation of the effects of icing on a rotor is, in general, more complicated than for a propeller. The propeller problem can be viewed as a special case of the rotor problem.

The analysis of propeller icing requires the determination of the local angle of attack and Mach number along the blade. For a propeller (at small aircraft angle of attack), the Mach number and blade angle of attack are not functions of rotational position. Photographs have not shown any evidence of ice beyond 99% of the span of the blade. Correlation work performed to date appears to substantiate the use of two-dimensional wind tunnel airfoil data in propeller calculations, although only limited data is available for correlation (reference 2-79).

2.3.3 Powerplant

Aircraft powerplants may suffer performance and/or physical damage due to icing in three broad categories:

1) Structural damage due to ice shedding.

2) Engine flow distribution causing stall and flameout.

3) Icing over of instrumentation necessary for engine operation.

For reciprocating engines, the amount of air intake is small and the internal icing problems are primarily related to carburetor icing. However, for gas turbines a large amount of air intake is required and, therefore, a large volume of supercooled water droplets is ingested. This sometimes leads to serious icing problems on the spinner for fans, or on the front bearing housing for non-fan, inlet guide vanes and the first row of compressor vanes and stator blades. Instrumentation for turbine engines is located toward the forward and aft parts of the engine to determine the pressure/temperature differential through the engine. This differential is extremely important since the power setting is based on these readings. The forward location can be susceptible to icing. It has been speculated that a malfunction of this gage could have been one of the factors in the disastrous

Perhaps the most serious fixed wing aircraft control problems due to ice accretion are those related to ice contaminated tailplane stall (ICTS). This phenomenon occurs when an iced horizontal stabilizer, carrying a download at a negative angle of attack, stalls during approach or landing. This may occur with few or no perceptible warning signals to the pilot, sometimes resulting in a large hinge moment and, in the most severe cases, a nose-down stick force ranging for one to several hundred pounds. A large difference in pressure between the top and bottom surfaces in the elevator region produces the large hinge moment and stick force. Most accidents and incidents attributed to ICTS have occurred on turbopropeller airplanes; this is believed to be due at least in part to the fact that turbopropeller commuters are likely to be exposed to icing conditions on a greater proportion of their flights than other airplanes. In addition, their options for avoiding icing conditions may be limited by route, altitude, or schedule. Approximately a dozen accidents and a large number of incidents attributed to ICTS occurred from the middle 1950s to 1992 (reference 2-U2). Seven airworthiness directives (ADs) for five turbopropeller airplanes were issued between 1982 and 1992. An international workshop on the subject, co-sponsored by the FAA and NASA, was held at NASA Lewis Research Center in 1991 (reference 2-U3).

There are several interlocking circumstances which make ICTS a particularly insidious safety problem.

1. Ice accretion may be more severe on the tailplane than on the wing.

   As noted earlier, the tailplane is a more efficient collector of ice than the wing because it has a smaller chord and because ordinarily it is thinner and has a smaller radius of curvature at its leading edge. As a result, the ice accretion on a tailplane is larger (relative to chord size) than that on a wing experiencing approximately the same icing conditions. In fact, the maximum thickness of the ice on the tailplane may be greater than that on the wing in absolute terms (reference 2-U4).

   There is evidence that icing conditions at the tailplane may be quite different, and sometimes substantially more severe, than those at the wing, perhaps as a result of downwash, propeller slipstream, or other affects. Based on examination of the wing and tailplane surfaces after landing, there have been reports of a maximum ice thickness on the tailplane two or three times greater than that on the wing, and other reports of substantial ice on the tailplane when there was none on the wing and when none had been observed during flight. This phenomenon is not well understood, but it has been suggested that some of these observations may be due to a small temperature depression in the area of the tailplane, or, alternately, to enhanced heat transfer at the tailplane surface.

2. The criticality of an ice accretion may depend more upon its roughness and location than on its size. Moreover, critical ice accretions may be difficult to specify with the present state of knowledge.

The difficulty of identifying critical ice accretions is illustrated by occurrences such as the following: An aircraft crashed due to ICTS with a maximum ice thickness of approximately one inch on the tailplane whereas a few months later another aircraft of the same model, having experienced similar icing conditions, was reported to have landed without incident at the same airport on the same runway with a maximum thickness of approximately three inches. Furthermore, recent work has shown that the roughness from a very slight accretion can change the stall angle of attack of an airfoil by several degrees (reference 2-U5).

3. The pilot flying a turbopropeller airplane may be able to observe ice on the wing but rarely on the tailplane. This renders the limitations of ice detection probes or point sensors more critical for the tailplane than for the wing.

4. ICTS may occur with little or no warning, the first symptom being very substantial nose-down stick forces. Wing stall due to icing ordinarily occurs with more warning to the pilot, and development of extremely adverse flight conditions is not ordinarily so sudden.

Following flap extension, one or more of the following symptoms may signal tailplane stall or impending tailplane stall: (reference 2-U6):

- Elevator control pulsing, oscillations, or vibrations.
- Abnormal nose down trim change.
- Other pitch anomalies possibly resulting in pilot induced oscillations.
- Reduction of elevator effectiveness.
- Sudden elevator force change (control would move nose down if unrestrained) followed by uncommanded nose down pitch.
- Sudden uncommanded nose down pitch.

Note that the first four would probably not be observed if flying with autopilot and that, if not severe, might be attributed to causes other than ice on the tailplane.

5. Protection of the tailplane using pneumatic boots, which are used on most turbopropeller airplanes, can be more difficult than protection of the wings.

Many pilots do not activate the boots until the maximum ice thickness on the wings is at least one-quarter of an inch. (This assumes they are able to see the wings well enough to estimate the ice thickness; it is done to avoid "bridging." See Chapter III, Section 1.0.) However, the ice

thickness on the tailplane may be considerably greater. This suggests it might be preferable to de-ice the wing and tailplane independently, but the pilot ordinarily cannot see the tailplane. Recall also that ice may accrete on the tailplane when none accretes on the wing.

Turbopropeller airplanes which have powered control surfaces are not necessarily less susceptible to ICTS, but possess the means to manage it since a large hinge moment can be overcome by the elevator power system. On the other hand, pilots of turbopropeller airplanes which rely on aerodynamic forces to keep stick forces low should be very alert to the symptoms and dangers of ICTS. High efficiency flaps that produce relatively high downwash and thus large negative angles of attack at the tailplane can increase the danger of ice ICTS occurring. Non-trimmable stabilizers and efficient stabilizers with short chord length and small leading edge radius also may increase the danger of ice CTS (reference 2-U4).

The most complete technical discussion of ICTS can be found in the work of Ingelman-Sundberg and Trunov (references 2-77, 2-93, and 2-U2) and much of the explanation that has been given here is based upon their work. Reference 2-77 reports on a wind tunnel study conducted on three-dimensional tailplane models (which included elevators) with simulated ice shapes. First, tailplanes using a NACA 64A-009 airfoil section were used. This section has a very small leading edge radius and its aerodynamic performance was not seriously affected by the ice accretions. These tailplanes were then modified by changing the leading edge to simulate a NACA 0012. Figure 2-117 shows the lift and elevator hinge moment coefficients for this tailplane configuration and several ice simulations. With the modified leading edge, the tailplane suffers a large $C_{L_{max}}$ penalty, and, near stall, a large nose down change in hinge moment coefficient occurs. This indicates that if a tailplane airfoil section is optimized for good $C_{L_{max}}$, ice can have a severe effect on aircraft longitudinal control when the tailplane must operate at large negative lift coefficients (large downloads).

As can be seen from figure 2-117 a large and sudden change in C_H occurs when the tailplane with ice, begins to stall. Wing flaps aggravate this situation by changing the downwash field at the tail, thus changing the required elevator deflection to trim the aircraft.

Trunov and Ingelman-Sundberg (references 2-93 and 2-U2) distinguish three "levels" of ice CTS following partial or full flap deployment: (1) The pilot experiences unusually large stick forces, but by holding back the yoke can prevent the aircraft from diving even without retracting the flaps. (Since the situation may worsen, the pilot should of course at least partially retract the flaps.) (2) The pilot is able to hold the yoke back so that the elevator is in full nose-up position, but the tailplane still cannot provide the needed download to maintain the desired trim speed. In this case, the pilot must at least partially retract the flaps. Since the dive should develop fairly slowly in this case, there will be sufficient time to do this. (3) The pilot is unable to prevent the yoke from going full forward. In the few seconds available to him, he must have the co-pilot retract the flaps while he pulls back as hard as possible on the yoke and then have the co-pilot assist with the yoke.

The problem of horizontal tail icing should be considered during aircraft design. The work of

Trunov and Ingelman-Sundberg suggests that aircraft least sensitive to tail icing problems are those such that (1) the tail incidence is adjustable, (2) the tail $C_{L_{max}}$ is low due to the airfoil section used, or (3) the tail is designed not to require large C_Ls. Flap deflection should be carefully analyzed or tested for its affect on iced tailplane performance. Many aircraft manufacturers suggest limiting flap deflection on aircraft when tailplane ice is suspected and, as noted above, ADs have been issued requiring certain aircraft to limit flap deflection during and after flight in icing conditions.

Karlsen and Sandberg (reference 2-94) have performed a simulation of aircraft longitudinal stability with tailplane ice. They found a lack of pitch response to strong downward vertical gusts and unstable oscillations in pitch when elevator control is applied to correct for glideslope tracking errors.

Ice may affect the aircraft stability and control in other ways. Ice accretion on the vertical surface could affect the rudder performance. This would be most likely to occur when maximum rudder power is needed in an engine-out case.

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Since the overall degradation of aircraft performance with ice has not been discussed, some brief comments will be made here. One of the earliest experimental studies of the performance of aircraft with ice accretion is that of Preston and Blackman (reference 2-82). They used selective de-icing of the various aircraft components to determine the drag increment due to each. In figure 2-118 the percent drag increase due to these components is shown. This research was conducted using a B-25 aircraft. It is important to note that a large percentage of the drag rise is from non-lifting surfaces. Unlike wing sections, little data is available on the drag rise due to ice accretion on these components.

In Leckman's 1971 paper (reference 2-95), he presents a method for predicting the effect of ice on subsonic aircraft performance. He demonstrates the procedure for a Cessna Centurion and a Super Skymaster. First, ice shapes are estimated using impingement data, then the drag penalties are estimated and a drag build-up procedure is performed. Table 2-10 gives the percent drag increase for the various components.

Leckman used experimental data from the NASA Icing Research Tunnel to estimate the drag increase of the flying surfaces. Table 2-10 represents a continuous maximum icing encounter where T = 17 °F, MVD = 20 μm, LWC = 0.46 g/m^3 and the icing encounter was 20 miles in length. The percentages above correspond to a total drag increase of ΔC_D = .0550 for the Centurion and ΔC_D = .0630 for the Super Skymaster. These calculations are compared to experimental data for the Super Skymaster. Performance with maximum continuous ice is compared to natural icing flight test results in figure 2-119 (reference 2-95).

Other studies have used similar methods to predict aircraft performance degradation with ice accretion. While the basic method is the same as Leckman's, a component build-up method for the drag, these studies computerized the procedure (references 2-96 and 2-97). These programs also made use of the drag correlations discussed earlier to predict wing and empennage drag rise. These codes, therefore, suffer the same inaccuracies as the correlations, but do provide an easy way to predict trends. A major problem is the estimation of the drag from the miscellaneous items as in table 2-10. A large portion of the drag can come from nacelle, fuselage, antenna, landing gear, etc. This is especially true on a "dirty" airplane, i.e., one with a large amount of parasitic drag. At this time, the percentage drag increase due to these non-lifting components must be estimated on the basis of flight tests or previous experience.

Little flight test data was available in the open literature prior to 1983 which could be used to verify calculation methods. In that year, NASA Lewis initiated an icing flight test program using the NASA Icing Research Aircraft (a Twin Otter), which has generated much useful information (references 2-98 and 2-99). One part of the program has been to measure the effect of ice on aircraft performance. After accreting the ice in steady flight, the aircraft exited the clouds to conduct an aircraft performance flight test. Figure 2-120 shows the measured aircraft drag polar for various degrees of de-icing. Note the changed slope of the C_D versus C_L^2 curve with the aircraft completely iced. Also note the large portion of the drag penalty still remaining after the wings and empennage were de-iced. Propeller and engine inlet heaters were on at all times during the flight. This flight

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TABLE 2-1. DROPLET PARAMETERS FOR THE LANGMUIR D DISTRIBUTION

δ	Δv	Re	K	K_0
6.2	0.05	39.3	0.015	0.007
10.4	0.10	65.9	0.042	0.017
14.2	0.20	90.0	0.078	0.029
20.0	0.30	126.8	0.155	0.050
27.4	0.20	173.7	0.292	0.083
34.8	0.10	220.6	0.471	0.121
44.4	0.05	281.5	0.766	0.176

TABLE 2-2. CALCULATION OF $\overline{E}$ AND $\overline{\beta}(S=0)$

K_0	Δv	E	$E \cdot \Delta v$	$\beta(S=0)$	$\beta(S=0) \cdot \Delta v$
0.007	0.05	0.02	0.001	0.25	0.013
0.017	0.10	0.11	0.011	0.44	0.044
0.029	0.20	0.17	0.034	0.56	0.111
0.050	0.30	0.23	0.069	0.68	0.203
0.083	0.20	0.32	0.065	0.76	0.152
0.121	0.10	0.40	0.040	0.82	0.082
0.176	0.05	0.49	0.025	0.86	0.043
			$\overline{E} = 0.24$		$\overline{\beta}(S=0) = 0.65$

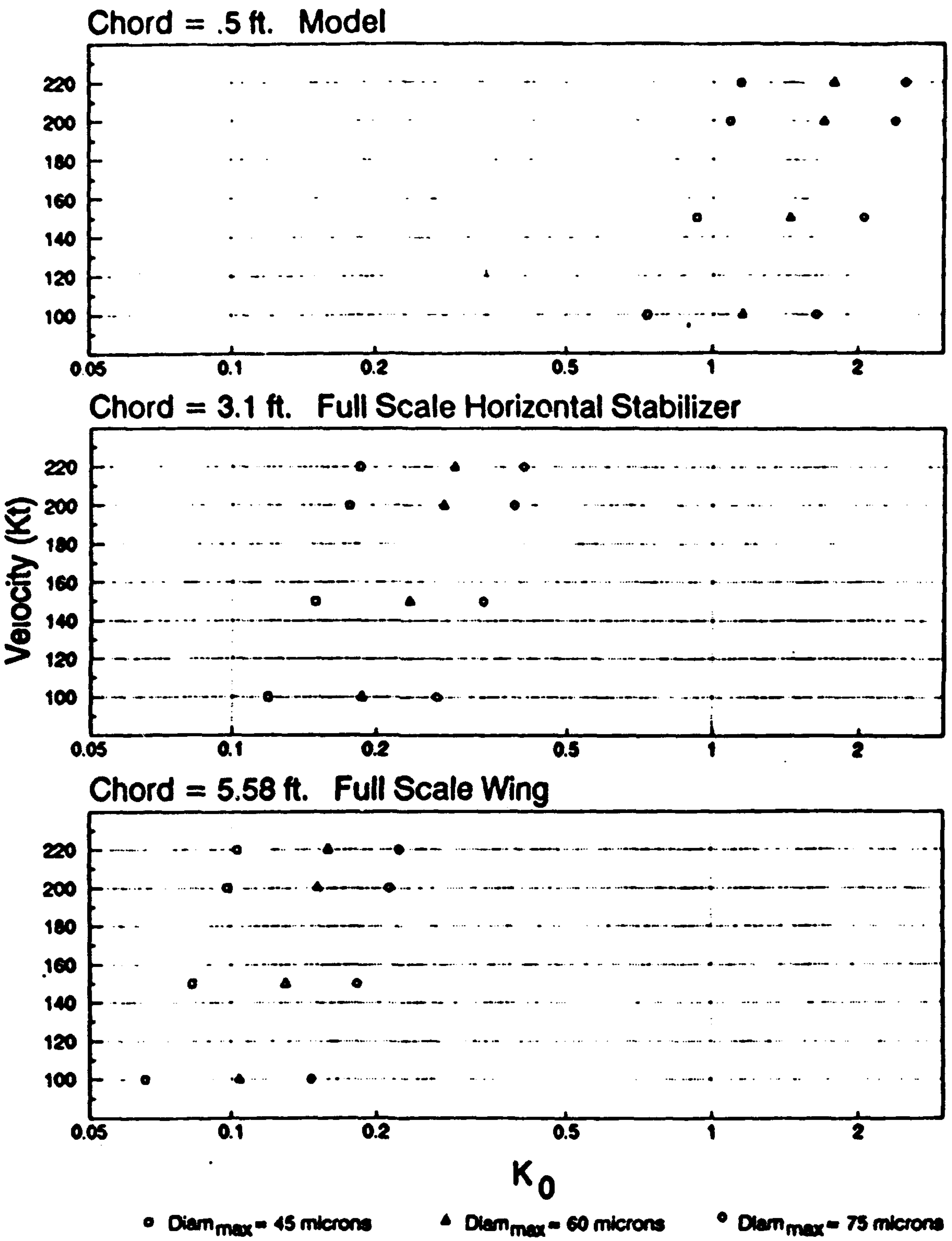

FIGURE 2-5. K_0 BASED ON DIAM$_{MAX}$ FOR SEVERAL CHORD SIZES

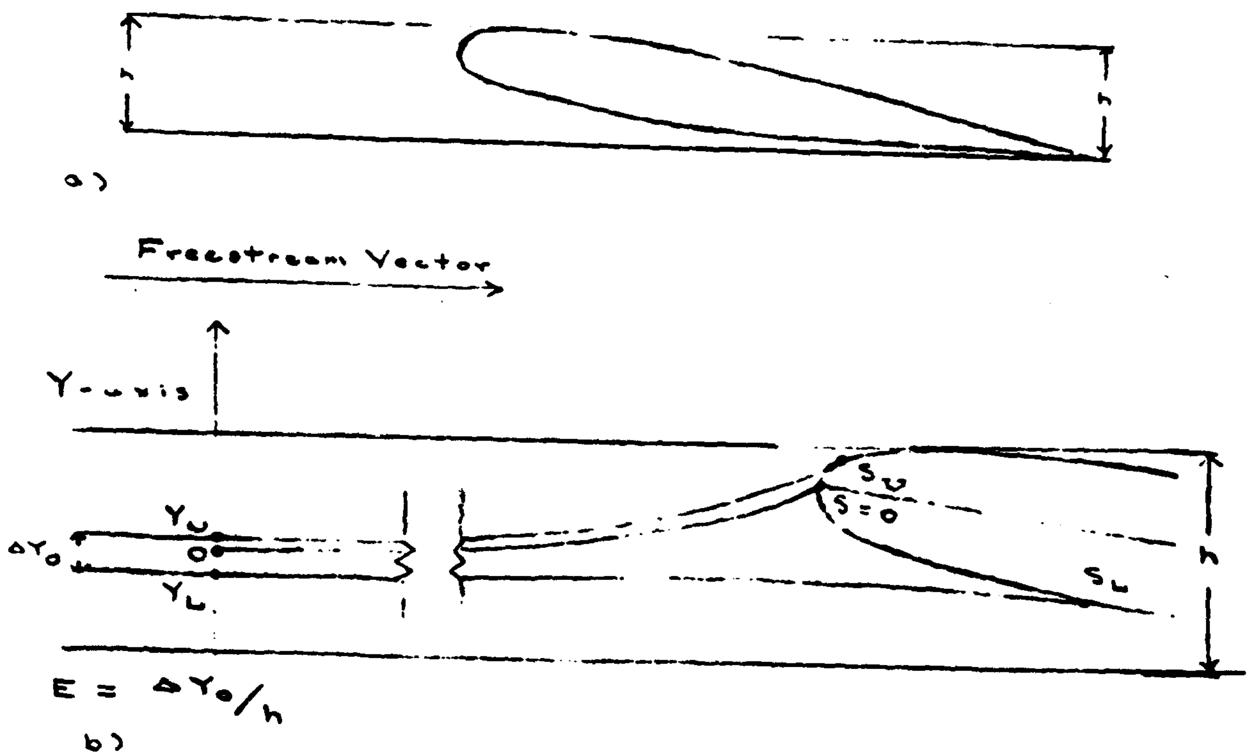

FIGURE 2-6. DEFINITION OF DROPLET IMPINGEMENT PARAMETERS

CHAPTER III

ICE PROTECTION METHODS

CHAPTER III
SECTION 1.0
CONVENTIONAL PNEUMATIC BOOT DE-ICING SYSTEMS

CHAPTER III - ICE PROTECTION METHODS

CONTENTS

SECTION 1.0 CONVENTIONAL PNEUMATIC BOOT DE-ICING SYSTEMS

LIST OF FIGURES

SYMBOLS AND ABBREVIATIONS

Symbol	Description
°C	Degrees Celsius
cm	Centimeter
°F	Degrees Fahrenheit
FAA	Federal Aviation Administration
ft	Feet or foot
gpm	Gallons per minute
HP	Horsepower
kg	Kilogram
kN	Kilonewton
lbf	Pounds force
lbs	Pounds
m	Meter
mm	Millimeter
psig	Pounds per square inch gauge (pressure)
scfm	Standard cubic feet per minute

GLOSSARY

bridging - The formation of an arch of ice over a pneumatic boot on an airfoil surface.

icephobic - A surface property exhibiting a reduced adhesion to ice; literally, "ice-hating."

light icing - The rate of accumulation may create a problem if flight is prolonged in this environment - over 1 hour. Occasional use of deicing/anti-icing equipment removes/prevents accumulation. It does not present a problem if the deicing/anti-icing equipment is used.

moderate icing - The rate of accumulation is such that even short encounters become potentially hazardous and use of deicing/anti-icing equipment or diversion is necessary.

III.1.0 CONVENTIONAL PNEUMATIC BOOT DE-ICING SYSTEMS

III.1.1 OPERATING CONCEPTS AND COMPONENTS

Pneumatic boot systems have been the standard ice protection method for piston engine aircraft since the 1930's. The boot surfaces remove ice accumulations mechanically by alternately inflating and deflating tubes within a boot that covers the surface to be protected. Inflation of the tubes under the accreted ice breaks the ice into particles and destroys the ice bond to the surface. Aerodynamic forces, and centrifugal forces on rotating airfoils, then remove the ice. This method of de-icing is designed to remove ice after it has accumulated rather than to prevent its accretion on the surface; thus, it cannot be used as an anti-icing device.

Conventional pneumatic boots are constructed of fabric-reinforced synthetic rubber or other flexible material. The material is wrapped around and bonded to the leading edge surfaces to be de-iced on wings or empennage. Total thickness of typical pneumatic boots is usually less than 0.075 inch (1.9 mm). Pneumatic boots are easily retrofitted, require very little power, and are a light weight system of reasonable cost.

The tubes in the pneumatic boot are usually oriented spanwise but may be oriented chordwise if dictated by a particular design. When inflated, chordwise tubes have lower drag than spanwise tubes but may present manifolding complications. The inflatable tubes are manifolded together in a manner to permit alternate or simultaneous inflation as shown in figures 1-1 and 1-2, but alternate inflation is less commonly used. Chordwise, the extent of de-icing coverage should be determined by analysis or test of droplet impingement limits (Section I.2.2.1.6). Spanwise coverage should be sufficient to protect the surface in question.

In addition to the boots, the primary components of a pneumatic system are a regulated pressure source, a vacuum source, and an air distribution system. Miscellaneous components may include check and relief valves, air filters, control switches and timer, and electrical interfaces including fuses and circuit breakers. A regulated pressure source is required to insure expansion of all tubes in the system to design limits and within design rise times. If tube expansion is too slow, de-icing effectiveness is lessened. The vacuum source is essential to insure positive deflation and keep the tubes collapsed during non-icing flight conditions to minimize the aerodynamic penalty.

Air pumps generally multiply the atmospheric pressure by a fixed factor, so the pressure delivered becomes a function of altitude. Therefore, for air pump systems, the pressure produced at service ceiling altitude is a design condition.

Some characteristics of a conventional pneumatic boot system are listed below:

Surface Ply Elongation	40 to 50 %
Nominal Inflation Time	Five seconds
Nominal Deflation Time	Six seconds
Maximum Surface Distortion	0.375 in. (9.53 mm)
Threshold Ice Removal Thickness	0.25 in. (6.35 mm)
Surface Ply Material	Elastomeric

A new pneumatic boot design has recently been developed that removes thin ice (down to 0.06 inches) on thin airfoils. The boot uses de-icing tubes that are a fraction of the size of conventional boot tubes and are inflated by higher air pressures for less than one second.

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III.1.2 DESIGN GUIDANCE

1.2.1 Fixed Wing Aircraft

Boot de-icing is strongly affected by the airfoil shape. The boot manufacturer's assistance is usually needed in the determination of tube size, sequencing order, pressure level, spanwise/chordwise tube combinations, and other attributes.

The system should be operated to evaluate overall performance during dry air flight testing. Boot inflation pressure should reach the design pressure within the allowable inflation time (usually about five or six seconds). This pressure should be maintained up to the maximum icing altitude of 22,000 feet (6100 m) (see FAR 25, Appendix C) or the aircraft's service altitude, whichever is lower. Also, the vacuum used to deflate the boots should be adequate even at maximum operating airspeeds.

1.2.1.1 Turbine Engine Powered Aircraft

Gas turbine engines generally provide pressure directly from compressor bleed air and vacuum from a bleed air driven ejector.

Components of a typical pneumatic boot surface de-icing system for a twin-turbine powered aircraft are shown schematically in figure 1-3. This typical system utilizes engine bleed air for the air pressure source, which is regulated to 18 lb/in^2 (124 KN/m^2) for boot inflation. As a safety feature, a relief valve is incorporated into the regulator valve design that will limit the over-pressure. For the dual cycle system shown, the wing and empennage boots may be alternately pressurized.

The regulated bleed air is routed to a venturi air ejector which provides vacuum for boot hold-down, as well as for flight instruments. A distributor valve applies pressure or vacuum to the boots in conformity with a selected cycle. Usually this valve has two boot distribution ports - one port is used to inflate and deflate the wing boots and the other port is used for the empennage boots. An alternate distributor valve has a single inflation port and incorporates an ejector for vacuum. Air plumbing line sizes and system components are selected based on the functional requirements; namely,

maximum boot operating pressure and the pressure rise time. Installation of this type system requires only minor airframe modifications.

1.2.1.1 Reciprocating Engine Powered Aircraft

For piston engine aircraft, air pumps driven from the engine's geared accessory drive are usually used. Some manufacturers use the inlet and outlet sides of a sliding vane air pump to provide both vacuum and pressure. If only the outlet side of the air pump is used, a dual-pressure regulator and control valve are necessary to supply low pressure air to the flight instruments and to an air injector for de-icer system vacuum, and also the higher pressure air required for boot inflation. Engine manifold vacuum is not suitable due to its extreme variability with engine load, and with turbo-charged engines, no manifold vacuum exists. Vacuum systems are often shared with vacuum-driven flight instruments.

Components of a typical pneumatic boot surface de-icing system for a reciprocating engine powered aircraft with positive air pressure flight instruments are shown in figure 1-4. Engine driven dry air pumps supply air pressure for boot inflation. Dual pressure regulator and relief valves control the pressure at a low pressure setting that is adequate for instrument operation. When the surface de-icing system is activated, the dual pressure regulators shift to the higher pressure required for pneumatic boot inflation. This two-stage pressure control provides extended pump life and less engine power extraction in normal flight without icing conditions. A timer operates the solenoids in the pressure regulators and the de-icing valve. A pressure switch operates a signal lamp to show boot operation.

The pressure regulator and relief valve system maintains pressure when the de-icing system is in use. The de-icing valve is a solenoid operated ON-OFF valve which applies pressure or vacuum to the boots. An air ejector is included in the system to provide vacuum to the boots in the OFF valve position. A single-cycle system where all boots are pressurized together is shown in figure 1-4.

1.2.2 Rotorcraft

An experimental pneumatic boot de-icing system has been successfully tested on helicopter rotor blades (references 1-1 and 1-2). A de-icing boot configuration was developed (figure 1-5) to minimize aerodynamic drag when the boot was inflated. In this test, the inflated boots caused a drag increase equivalent to about 3/8 inch (.95 cm) ice on the rotor blades. For a 9500 lb (4310 kg) 2-blade helicopter, full span de-icing boots were simultaneously inflated in less than two seconds to effectively remove accreted ice. Operating air pressure was obtained from a turbine engine bleed source. Figure 1-6 shows the operating schematic of this system. Improved ice shedding indicated that the boot rubber surface had a reduced surface adhesion to the ice.

1.2.3 Other Applications

Ice protection of some other components, such as radomes, with pneumatic boot de-icing systems is feasible (see Section 1.3.8).

III.1.3 USAGES AND SPECIAL REQUIREMENTS

1.3.1 Airfoil and Leading Edge Requirements

The airflow required for pneumatic boot operation is small compared with that for a hot gas ice protection system. Pneumatic boot de-icing systems may be added to an existing airplane with minor modification and expense. In areas of low static pressure on airfoils, auto-inflation of pneumatic boot tubes may occur and disrupt airflow over the surface. A vacuum source is used to prevent auto-inflation during the deflation period. During the inflation portion of the cycle, large drag increases and lift degradation can occur because of the spoiler action of inflated spanwise de-icing tubes. The use of chordwise tubes minimizes this problem.

Ice particles shed by pneumatic boots may be large enough to damage aft-mounted engines or propellers. Axial flow engines (turbojets and turbofans) are the most vulnerable, while turboprop engines with particle-separating inlet ducts are less likely to be damaged. For some airplanes, the wing section upstream of the engine may be provided with some form of anti-icing to avoid engine ice ingestion while the remainder of the wing is de-iced by pneumatic boots. The silver pneumatic boots have the capability of removing thinner ice accretions (1/8") which will not damage rear-mounted pusher propellers. Such a system has been certified on aft-mounted propeller aircraft.

1.3.2 Windshields

The application of pneumatic boots for windshields is not possible.

1.3.3 Engine Inlet Lips and Components

The use of pneumatic boots has been limited to ice protection of turbine engines with bypass inlets.

1.3.4 Turbofan Components

The use of pneumatic boots for turbofan components has not been tried.

1.3.5 Propellers, Spinners, and Nose Cones

The use of pneumatic boot de-icing on propellers, spinners, and nose cones is feasible but has not been tried.

1.3.6 Helicopter Rotors and Hubs

Pneumatic boot de-icing systems have been tried on helicopters on an experimental basis using a 9500 lb (4310 kg) 2-blade helicopter (references 1-1 and 1-2) as discussed in Section 1.2.2. No application to rotor hubs is known.

1.3.7 Flight Sensors

Pneumatic boot systems are not suitable to de-ice flight sensors.

1.3.8 Radomes

Radar-designed pneumatic boot de-icers may be installed on the external contour of radomes; however, the boot may slightly increase transmission losses. A schematic of a pneumatic boot system applied to a radome is shown in figure 1-7. This system inflates all tubes at the same time. A combination valve provides deflation vacuum or inflation air as determined by a control timer. Installation details are shown in figure 1-8. The boot is about 0.075 inches (1.91 mm) thick except in the supply manifold area where it is 0.16 inches (4.1 mm) thick.

1.3.9 Miscellaneous Intakes and Vents

Flush or recessed air scoops may not require ice protection. Pneumatic boot de-icing of air intakes and vents may be feasible, depending on the size of the intake or vent, but no application is known.

III.1.4 WEIGHT AND POWER REQUIREMENTS

The weight of a pneumatic boot ice protection system for a small, single-engine, FAR Part 23 airplane is approximately 25 lb (11 kg) and requires about one-third horsepower (250 watts), intermittently. The distribution of the system weight should not significantly affect aircraft balance and the total weight should not cause an appreciable performance penalty. The power extracted to drive an air pump in a piston engine powered aircraft is small in relation to the total power available.

For a small twin-engine FAR Part 23 airplane, a pneumatic boot ice protection system will weigh approximately 28 lb (12.5 kg) and require about one-half HP (370 watts), intermittently. The distribution of system weight should not significantly affect aircraft balance and the total weight should not cause an appreciable performance penalty.

For a small twin jet engine FAR Part 25 business jet airplane, a pneumatic boot ice protection system will weigh approximately 35 lb (16 kg) and require about one-half HP (370 watts), intermittently. For a large FAR Part 25 transport category airplane, the system will weigh approximately 195 lb (90 kg) and require about 2.8 HP (2100 watts).

For a 9500 lb (4310 kg) FAR Part 27 helicopter, a pneumatic boot de-icing system (figure 1-6) will weigh approximately 40 lb (18 kg). The weight breakdown of this system: is inflatable boots 22 lb (10 kg), components 3.8 lb (1.7 kg), and plumbing 14.8 lb (6.7 kg). Operating air for a two-second inflation cycle is about 22 ft^3 per minute. Electrical power required for this cycle is about one-half HP (370 watts), intermittently. For a larger FAR Part 29 transport category helicopter, the system weight and power required would be in proportion to aircraft weight.

III.1.5 ACTUATION, REGULATION, AND CONTROL

A pneumatic boot de-icing system is usually controlled by a three-position switch with OFF, MANUAL, and AUTO CYCLE modes of operation. When the switch is actuated in the MANUAL position, the de-ice system will operate through one cycle and return to the OFF position.

III.1.6 OPERATIONAL USE

Preflight checkout of the pneumatic boot de-icing system pressure and boot inflation is recommended. Generally, a nominal ice thickness of 0.5 inches is allowed to accrete before the de-ice system is turned on. Bridging is the formation of an arch of ice over the boot which is not removed by boot inflation. This can occur if the system is activated too early or too frequently, especially in glaze icing conditions. As icing encounters and severity of icing are difficult to forecast, the pilot should not depend upon marginal reserve power when ice protection is required and fly into an area where icing is predicted. Operation of a pneumatic boot ice protection system in ambient temperatures below -40 °F (-40 °C) may lead to permanent damage to the de-icing boots.

Pneumatic boots should inflate and deflate rapidly to function effectively. To accomplish this, the time to reach full pressure should be about 5 to 6 seconds.

In tests to date on rotorcraft, the pneumatic boot system is activated when ice growth reaches approximately a 0.25-inch (6 mm) thickness or when the indicated torque increases noticeably above the level with no ice accretion. Rotorcraft typically have smaller airfoils chords than fixed wing aircraft, so thick ice will result in high rotor power penalties, also thick ice may self-shed asymmetrically. The boot inflation time is approximately 2 seconds in rotorcraft applications.

An ice detection light is usually installed where it will illuminate a wing leading edge surface as an aid in observing ice accumulation during night operation. The location of the light and area illuminated must be such that the pilot can readily observe ice accretion and its thickness.

Liquids that reduce ice adhesion (icephobic) are available for applying to boots prior to a flight when an icing encounter is likely. These sprays reduce the adhesion of ice to the boot surface resulting in improved de-icing. However, the liquid erodes away so that it must be replenished after 50 to 150 flight hours.

III.1.7 MAINTENANCE, INSPECTION, AND RELIABILITY

Because pneumatic boot de-icing systems operate on clean turbine engine bleed or filtered air from dry air pumps, little is required in servicing the system. All vacuum and pressure filters used in the system should be periodically cleaned. Frequency of this cleaning will vary with the conditions under .vhich the airplane is operated.

The pressure regulating valves in the system ordinarily should not require adjustment although the valve assembly will usually be equipped with adjusting screws to permit field adjustments.

The dry air pumps require no lubrication or maintenance but should be overhauled or replaced at engine overhaul.

Surfaces of the pneumatic boots should be inspected for engine oil after servicing and at the end of each flight. Any oil deposits should be removed with non-detergent soap and water solution. Care should be exerciseu during cleaning to avoid scuffing the boot surface. Pneumatic boots may he damaged if refueling hoses are dragged over the surface of the boots, or if ladders and platforms are rested against them. In any event, the boot manufacturer's recommendations should be followed for maintenance and repair of cuts and scuff damage.

III.1.8 PENALTIES

Some aerodynamic drag penalty is to be expected with pneumatic boot de-icing systems on an airfoil but it can be lessened by recessing the surface leading edge to offset the boot thickness, or eliminated by a molded de-icer/composite leading edge assembly.

III.1.9 ADVANTAGES AND LIMITATIONS

Pneumatic boot de-icing systems have been in use for many years and their repair, inspection, maintenance, and replacement are well understood (references 1-3, 1-4 and 1-5). System weight and power requirements are minimal. Pneumatic boot material deteriorates with time and periodic inspection is recommended to determine need for replacement.

III.1.10 CONCERNS

A certain degree of pilot skill is required for safe and effective pneumatic boot operation. Actuation when accreted ice is too thin may result in "bridging" where the formation of ice over the boot is not cracked by boot inflation. Thus, attention is required to judge whether the cycle time continues to be correct as icing conditions change. Demands on the pilot increase during flight in darkness since observation of ice accretion rate and severity is more difficult.

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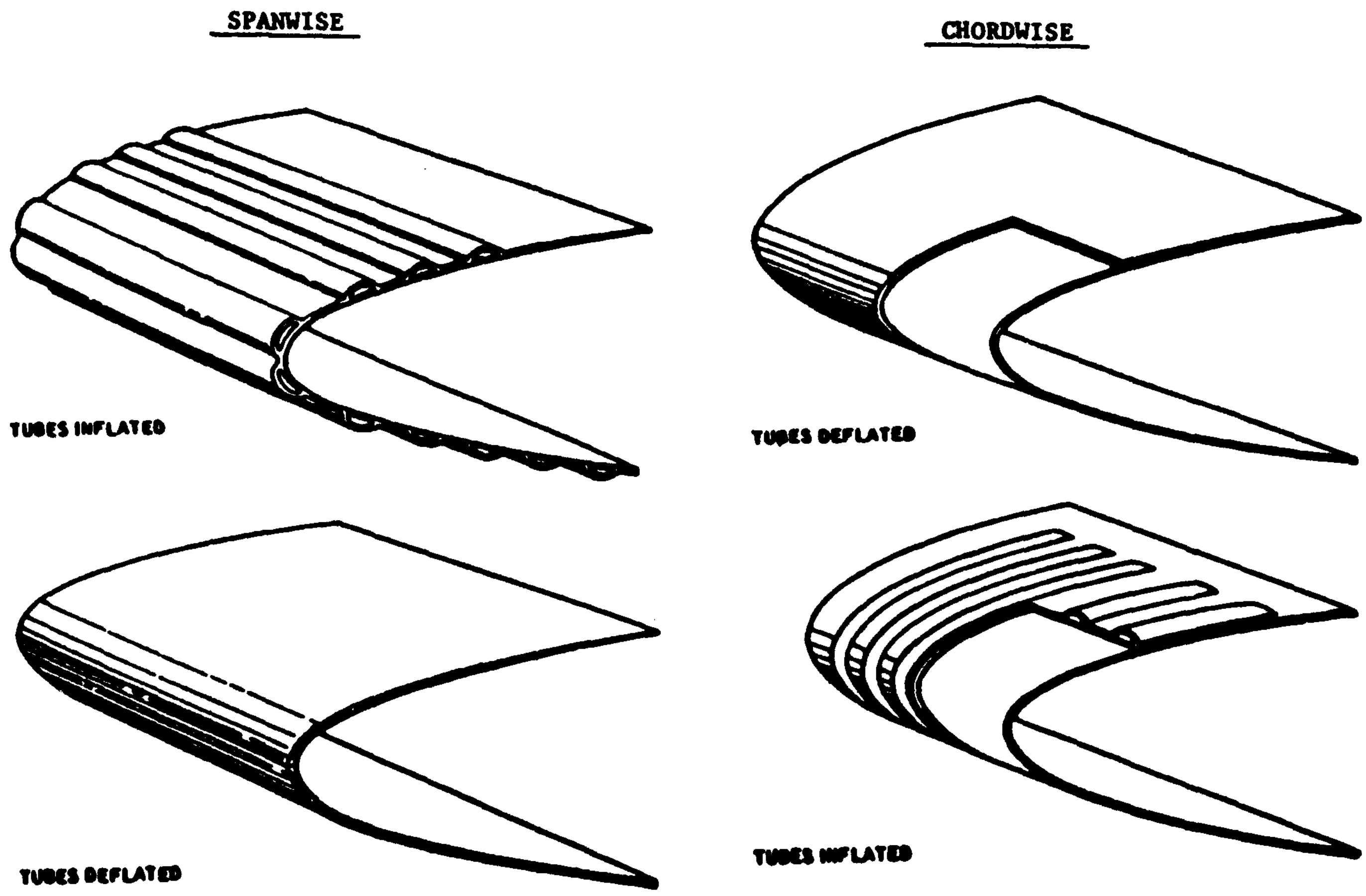
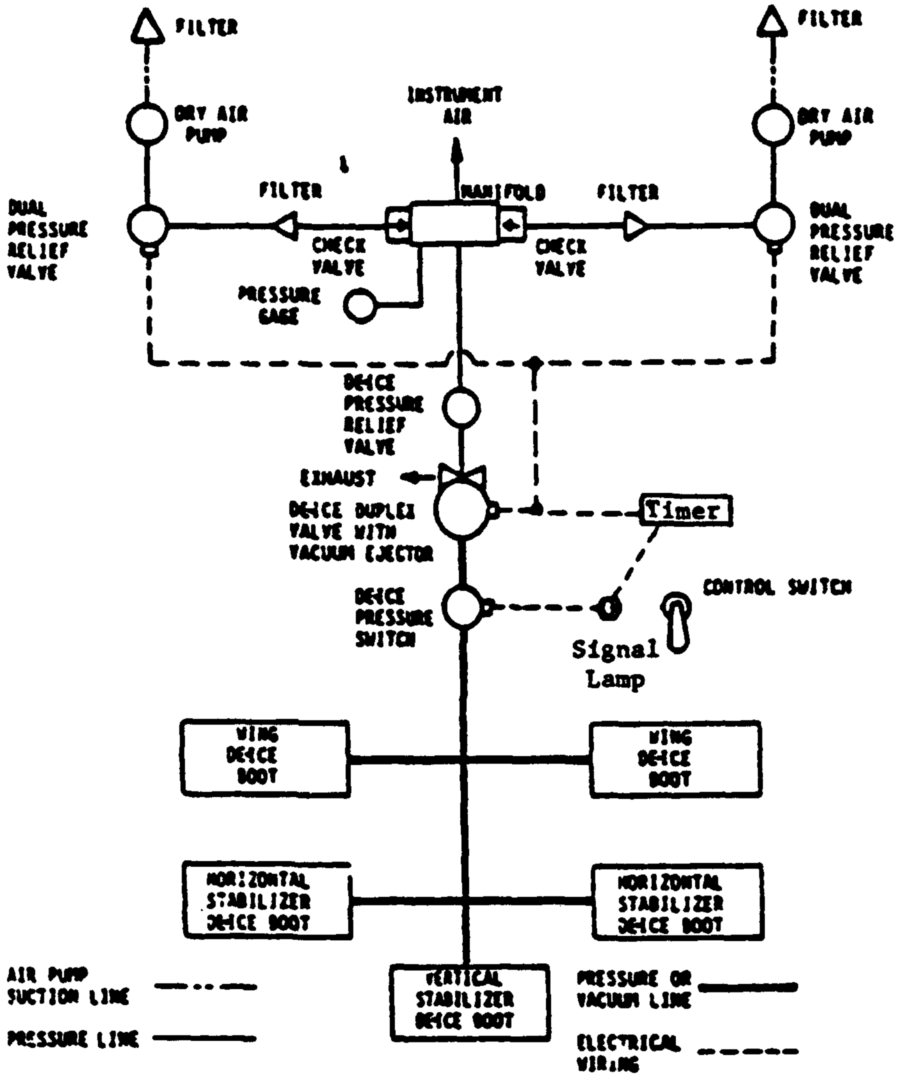


FIGURE 1-1. INFLATABLE DE-ICING TUBES



Single Cycle System

FIGURE 1-4. PNEUMATIC BOOT SURFACE DE-ICING SYSTEM - TWIN RECIPROCATING ENGINE POWERED AIRCRAFT

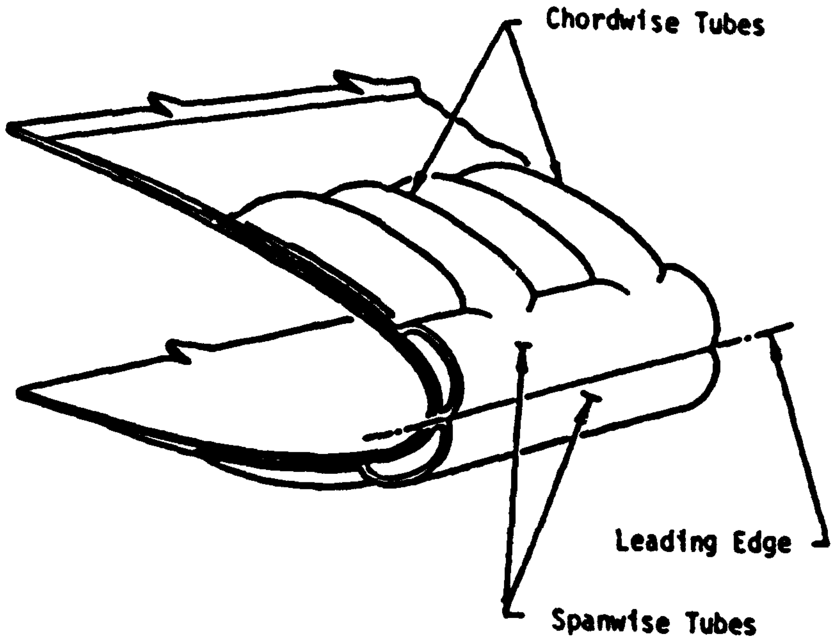


FIGURE 1-5. ROTORCRAFT BLADE PNEUMATIC BOOT

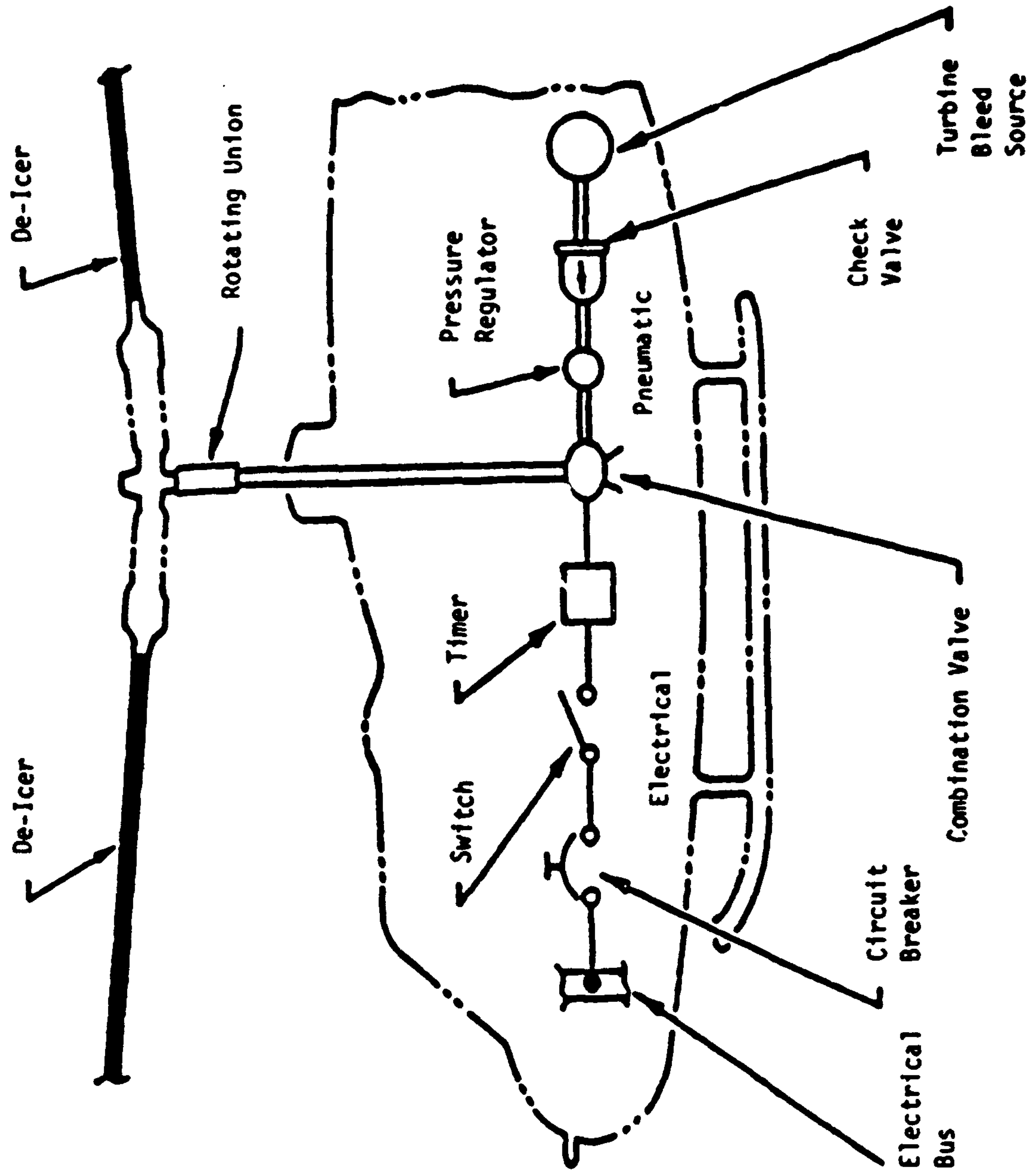


FIGURE 1-6. ROTORCRAFT PNEUMATIC BOOT DE-ICING SYSTEM - SCHEMATIC

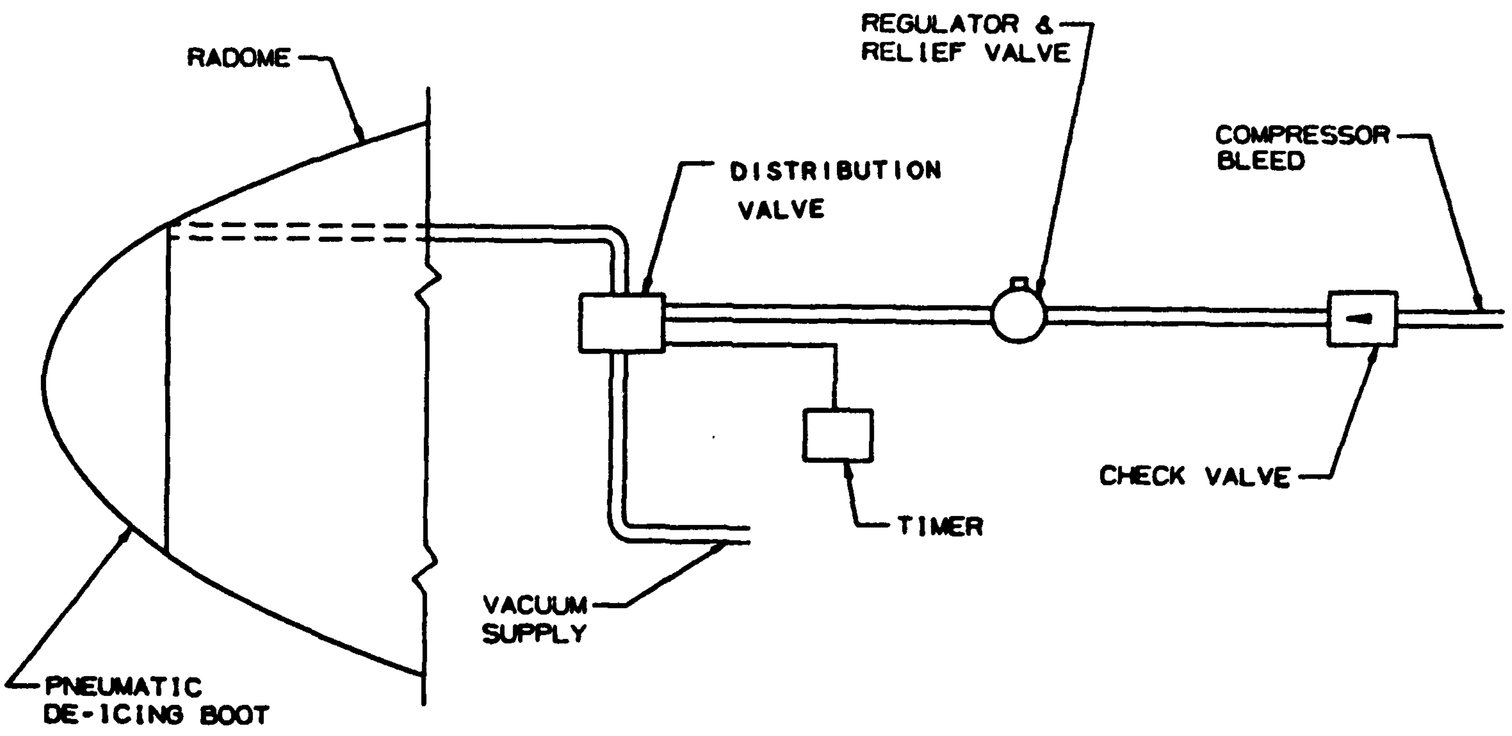


FIGURE 1-7. PNEUMATIC BOOT DE-ICING SYSTEM - NOSE RADOMES

CHAPTER III

SECTION 1A.0

PNEUMATIC IMPULSE DE-ICING SYSTEMS

CHAPTER III - ICE PROTECTION METHODS

CONTENTS

SECTION 1A.0 PNEUMATIC IMPULSE DE-ICING SYSTEMS

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LIST OF FIGURES

SYMBOLS AND ABBREVIATIONS

Symbol	Description
°C	Degrees Celsius
cm	Centimeter
°F	Degrees Fahrenheit
ft.	Feet
gpm	Gallons per minute
Hp	Horsepower
in.	Inch
KPa	Kilopascals
lbf	Pound-force
lbm	Pound-mass
LWC	Liquid Water Content
mm	Millimeter
m	Meter
MPa	Megapascals
OAT	Outside Air Temperature
PEEK	Polyetheretherketone
PIIP™	Pneumatic Impulse De-icing
psig	Pounds per square inch-gauge
psia	Pounds per square inch-absolute
SCF	Standard Cubic Foot
SCFM	Standard Cubic Feet per Minute
sec	Seconds
VDC	Volts - Direct Current
w	Watts

GLOSSARY

equivalent spherical diameter - The uniform diameter an ice shard would have after melting into a liquid water droplet.

icephobic - A surface property exhibiting a reduced adhesion to ice, literally, "ice-hating".

III.1A.0 PNEUMATIC IMPULSE DE-ICING SYSTEMS

III.1A.1 OPERATING CONCEPTS AND COMPONENTS

Pneumatic Impulse De-Icing, also called Pneumatic Impulse Ice Protection (PIIP™) is a mechanical ice removal system, which fractures, debonds and expels accreted atmospheric ice from the ice-accumulating surfaces of aircraft.

The removal is accomplished by a rapid distortion of the outer surface which occurs upon the introduction of a controlled burst of expanding high pressure air into a collapsed flexible channel, or impulse tube, located underneath the surface (figure 1A-1). As the expanding air traverses through the tube, the overlying surface is "snapped" outward, inducing bending stresses in the surface and the attached ice, as well as shear stresses at the ice/surface interface. The removal process is augmented by the high outward normal velocity which is imparted to the surface by the expanding air, as well as by the airstream. The expanded air is then vented to ambient through ports located in the backside of the de-icer.

The system consists of one or more de-icers which cover the surface(s) to be protected. Typically, the de-icer is built into the leading edge or ice-accreting structure so as to form a smooth, aerodynamically non-intrusive surface. The de-icer consists of a thin erosion surface overlying a quasi-flexible polymeric matrix which houses the fabric-reinforced impulse tubes (figure 1A-1). The tubes are typically oriented in the spanwise (longitudinal) direction. The de-icer is designed so that the surface region of action of the impulse tubes, matches the region in which atmospheric ice may be expected to accrete for the particular application. This region is determined by analysis or testing of the limits of impingement, for the defined icing conditions.

The regulated high pressure air source may be either a dedicated on-board compressor, stored air reservoir or bottle, or other high-pressure pneumatic system. System operating pressure is typically 600 to 1200 psig.

One or more impulse valves deliver a metered quantity of high pressure air to the impulse tube(s) in the de-icer in a manner which achieves the desired movement characteristic, or "impulse," on the surface. Valve activation occurs upon a signal from the controller, typically 28 VDC, for 0.05 sec. duration. Upon activation, the regulated supply air to the valve is shut-off, and the impulse air is discharged into the de-icer. The quantity of air required per impulse is typically sufficient to de-ice up to 8 sq. ft. of surface area. Upon de-activation of the valve, regulated supply air is allowed to "recharge" the valve. "Recharge" time is typically one second. The controller provides the timing and switching functions of the 28 VDC control signals required for operating the impulse valves. The controller also provides the fault detection and annunciation functions for verifying proper operation of the system.

Ancillary system operating equipment includes:

- Regulator: Regulates source air to system operating pressure.
- Pressure Switch: Used to indicate delivery of a satisfactory impulse to the controller.
- Shut-off Valve: Used for enabling/disabling compressor hydraulic supply or reservoir supply line when system is turned ON/OFF.
- Air Supply Conduit: Typically size -4 (1/4 in. OD), 3000 psig (20.7 MPa) tubing for distribution of regulated supply air to the impulse valves.
- Wire Harness: Used for interconnecting controller to impulse valves and pressure switches for control and fault detection.

A typical system schematic is shown in figure 1A-2.

III.1A.2 DESIGN GUIDANCE

1A.2.1 De-Icer Embodiments

The surface material may be titanium alloy, the high performance thermoplastic called polyetheretherketone (PEEK), or a toughened, impact-resistant, fabric-reinforced thermoset resin composite. PEEK or composite surfaces are generally used for applications in which a metal surface is not desirable. All surfaces have been tested extensively for ice removal performance and cycle life.

The de-icer may be configured in a number of ways, depending on the manner in which it is desired to be installed on the aircraft:

- SKIN-BONDED: The de-icer is bonded to an existing leading edge skin, or structure, in a manner similar to conventional pneumatic de-icers. This method is most suitable to field installations or retrofit applications, and facilitates removal and replacement; however it results in an aerodynamically-intrusive installation if not recessed.
- RECESS-BONDED: The de-icer is bonded into a recess in the skin, resulting in a non-intrusive installation.
- INTEGRATED COMPOSITE LEADING EDGE ASSEMBLY : The de-icer is manufactured with composite structural backing, designed to meet the structural requirements of the application, resulting in a "stand-alone" composite leading edge assembly with the de-icing function built-in. This is the most desirable embodiment for an aerodynamically smooth and non-intrusive installation.
- MODULAR COMPOSITE LEADING EDGE ASSEMBLY (MCLEA): This is similar to the integrated composite leading edge assembly, except that the surface assembly of the ice protector, consisting of the surface and its composite reinforcement, is mechanically attached, rather than internally bonded, to the underlying portion of the ice protector and leading edge structure. This allows the surface assembly to be removed and

replaced as a separate item should the surface be damaged, without requiring replacement of the entire leading edge.

1A.2.2 Air Supply And Distribution

The system is powered by high pressure air provided by a dedicated on-board compressor, a stored air reservoir, or another high-pressure aircraft pneumatic system.

The compressor may be either hydraulic or electric motor-driven. The hydraulic option is generally more desirable from a weight and size standpoint, provided ample flow capacity from the aircraft hydraulic system is available. The stored air reservoir has the obvious disadvantage of requiring periodic recharging.

Generally, the high pressure air is regulated to system operating pressure, and distributed via a high pressure line to one or more impulse valves located in the vicinity of the surfaces to be protected. System operating pressure is 600 psig (4140 KPa) nominal for PEEK-surfaced and composite-surfaced de-icers, and 1200 psig (8280 KPa) nominal for the titanium-surfaced embodiment.

1A.2.3 Impulse Valve Location

It is generally preferable to locate the impulse valves as close to the region to be de-iced as possible, in order to minimize attenuation of impulse strength. In practice, the valves are located immediately behind the leading edge surface wherever possible, and are connected directly to the inlet ports which access the de-icing tubes. Where space does not permit this type of installation, it is possible to locate the valves remotely from the de-icer, with some reduction in the surface area that may be effectively de-iced. The air inlet ports are typically located on the backside of the de-icer, but may be provided externally for applications, such as rotor blades, where internal access is not possible.

III.1A.3 USAGES AND SPECIAL REQUIREMENTS

1A.3.1 Airfoil and Leading Edges

As with most mechanical ice removal systems, thicker ice will more readily be shed than thin ice. Thus the thinness of the ice which can be effectively removed becomes a measure of the ice removal performance of the system. In extensive icing tunnel tests, the de-icer has demonstrated maximum ice thicknesses of 0.080" to 0.100" in continuous cyclic shedding in all icing conditions, including difficult slush ice, with shed ice particles sizes less than 0.25 in. equivalent spherical diameter.

1A.3.2 Windshields

Pneumatic impulse de-icing is not applicable to windshield de-icing.

1A.3.3 Engine Inlet Lips and Components

PEEK-surfaced and composite-surfaced pneumatic impulse deicers have been demonstrated to provide effective ice removal performance on engine inlet lips. Use of titanium on surfaces with compound curvature has not yet been developed.

1A.3.4 Turbofan components

The suitability of pneumatic impulse for use on turbofan components has not yet been evaluated.

1A.3.5 Propellers, Spinners, Nose Cones

The suitability of pneumatic impulse for use on these surfaces has not yet been evaluated.

1A.3.6 Helicopter Rotors and Hubs

PEEK, composite-surfaced, or titanium-surfaced pneumatic impulse embodiments have all been tested extensively and satisfactorily on rotor-blade airfoil shapes, but in a fixed rather than rotating condition. Special consideration would have to be given to location of the impulse valves and de-icing ports, as well as the transmission of the high pressure source air through a rotating union. These items have not been addressed to date.

1A.3.7 Flight Sensors

Pneumatic impulse de-icing is not suitable for the protection of flight sensors.

III.1A.4 WEIGHT, POWER AND ENVELOPE REQUIREMENTS

The weight, power and size estimates listed in table 1A-1 are for guideline purposes only and may vary depending on the application. Also these values reflect the current state of the system and may change with continued operational experience. De-icer values are specified on a unit area basis, and discrete components are specified on a unit basis. Weight and power required for a specific application may be estimated by multiplying the appropriate values by the surface area which would be required to be de-iced.

III.1A.5 ACTUATION, REGULATION, AND CONTROL

The controller automatically and repetitively operates the impulse valves in a manner which provides symmetric shedding about the aircraft centerline, typically on a fixed time cycle basis. Generally the system may also be commanded to perform a single shedding cycle "on demand". System function may also be initiated by a remote signal, such as from an OAT, LWC, or ice

detector sensor. Typically, the control and fault detection functions are provided by a dedicated de-icing system controller, but these functions may also be assigned to an onboard computer.

III.1A.6 OPERATIONAL USE

Pre-flight checkout of the de-icing system, by use of a self-test mode, is recommended. The system is capable of operating in the temperature range of -67°F to +165°F. The system should be activated by selection of the AUTO cycle mode when icing conditions are known or expected. In this mode the system will cycle continuously on a predetermined, fixed-time basis, typically one-minute cycles, until the system is switched OFF. A momentary MANUAL command may also be used to operate the system for one cycle of the entire aircraft "on demand."

There is no minimum or maximum ice thickness required or recommended for initiation of system operation.

There are presently no reduced ice adhesion (icephobic) or weathering-enhancing coatings required or recommended with either the titanium or PEEK-surfaced de-icers.

Composite-surfaced de-icers may require a coating to withstand the effects of rain erosion, particularly for high-speed aircraft. The coating may be spray-applied in the field periodically as required to refurbish the surface.

Use of paint on active de-icing surfaces is not recommended unless approved by the aircraft manufacturer.

III.1A.7 MAINTENANCE, INSPECTION, AND RELIABILITY

Lack of operational and service experience precludes accurate estimates of maintenance intervals and reliability.

Periodic visual inspection of the de-icing surfaces is recommended for detection of foreign object damage or fatigue cracks. Impulse valves should be accessible for repair or replacement, as should the compressor and controller. Periodic filter replacement and lubricating oil changes, if applicable, may be required on the compressor. No routine maintenance is presently required on the impulse valve, the de-icer, or the controller.

III.1A.8 ADVANTAGES AND POTENTIAL TRADE-OFFS

Advantages of the system are:

a. Low power requirements.
b. Aerodynamically non-intrusive in an integrated composite leading edge embodiment.
c. Thin ice removal capability: 0.080" to 0.100" ice thickness in all icing conditions, and shed ice particle sizes less than 0.25" equivalent spherical diameter.
d. Low radar capability in PEEK-surfaced and composite-surfaced embodiments.
e. No runback and refreezing.

Potential trade-offs of the system are:

a. The system is not presently installed or certified on any aircraft. Therefore, field service data on maintenance and reliability is not available.

b. As with all mechanical de-icing systems, some residual ice will remain after cycling.

c. Noise associated with pulsing the system has to be considered.

d. Ordinarily the system must be designed into the leading edge, and therefore is not readily suitable for retrofit applications.

III.1A.9 CONCERNS

Fatigue of the de-icer surface is a concern, particularly in view of the lack of operational experience with the system. Laboratory testing to date has demonstrated over 250,000 impulse cycles prior to the onset of surface fatigue and efforts are continuing to reduce the stresses which are induced by impulse.

III.1A.10 REFERENCES

1A-1 Sweet, D., "Development of an Advanced Pneumatic De-icing System," A-87-46-65-J000, American Helicopter Society, 43rd Forum, May 1987.

1A-2 Leffel, K.; Putt, J.; Martin, C., "Development of an Advanced Impulse Deicing System," AIAA-89-0492, paper presented at the 27th Aerospace Sciences Meeting, Reno, Nevada, Jan. 1989.

1A-3 Ramamurthy, S.; Keith, T.; DeWitt, K., "Numerical Modeling of an Advanced Pneumatic Impulse Ice Protection System (PIIP) for Aircraft," AIAA-91-0555, paper presented at the 29th Aerospace Sciences Meeting, Reno, Nevada, Jan. 1991.

1A-4 Martin, C. A.; Putt, J. C., "Advanced Pneumatic Impulse Ice Protection System (PIIP) for Aircraft," J. Aircraft, Vol. 25, No. 4, July-Aug. 1992, pp. 714-716.

TABLE IA-1

System Weight, Power, and Size Estimates

ITEM	UNIT WEIGHT	QUANTITY or APPLICATION	POWER	ENVELOPE DIMENSIONS
De-Icer	0.50 lb/sq.ft.*	-	-	0.100" thick
Impulse Valve	1.0 lb.	1 valve/8 sq. ft.	300W***	5.5 x 4.2 x 1.5"
Compressor	21.0 lb.	one	1.6 gpm**	11.5 x 9 x 10.3"
Controller	1.0 lb.	one	3W	4.8 x 4.0 x 3.6"
Regulator	0.2 lb.	one	-	2.6 x 2.0 x 1.0"
Shut-Off Valve	0.8 lb.	one	24W	3.2 x 2.5 x 2.0"
Pressure Switch	0.2 lb.	one per impulse valve	-	2.5 x 1.0 x 1.0"

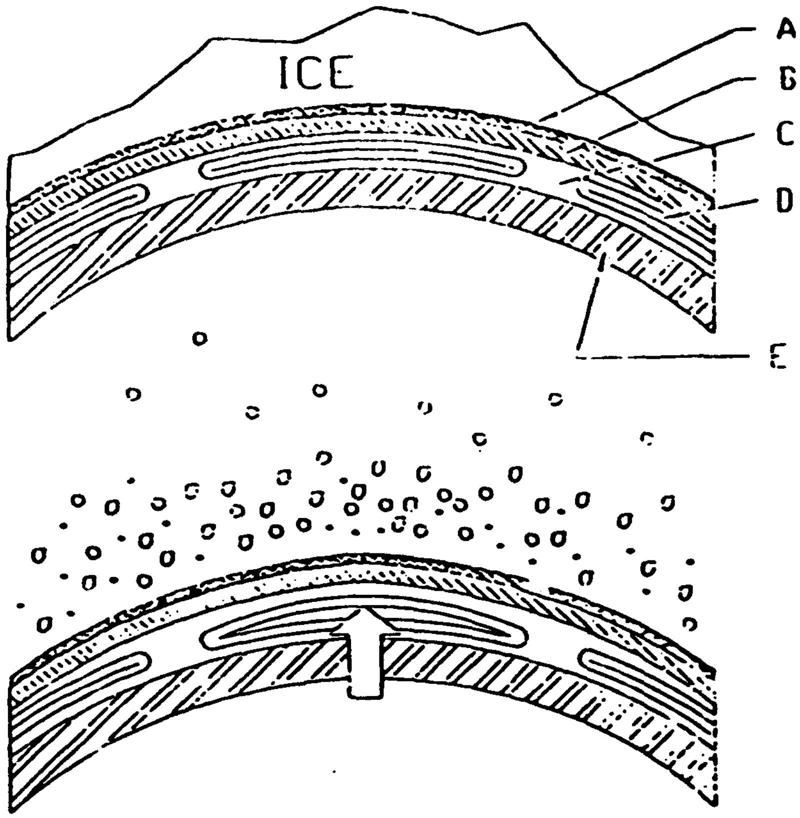
* per sq. ft. of ice accreting surface area.

** nominal flow rate of 2500 psig hydraulic fluid.

*** intermittent power requirement: 0.1% duty cycle.

Notes. Compressor weight and power values based on hydraulic motor-driven unit. Values for electric motor-driven unit will vary.

De-icer weight does not include composite substructure which must be designed in accordance with aircraft manufacturer's structural requirements.



A	SURFACE	TITANIUM
B -	SURFACE REINFORCEMENT	PHENOLIC/GRAPHITE
C -	MATRIX	NITRILE-PHENOLIC
D -	IMPULSE TUBE	NYLON
E	LEADING EDGE STRUCTURE	EPOXY/GRAPHITE

FIGURE 1A-1. TYPICAL INTEGRATED COMPOSITE LEADING EDGE ASSEMBLY

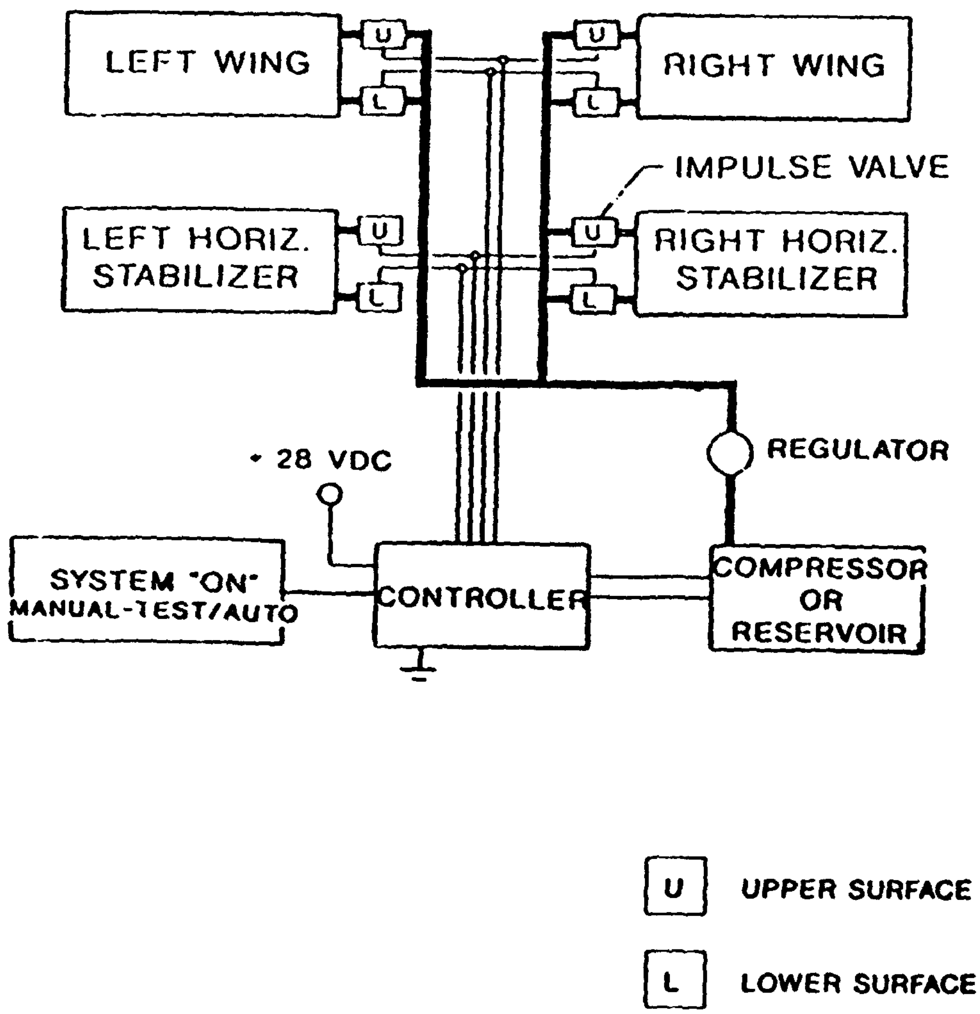


FIGURE 1A-2. TYPICAL SYSTEM SCHEMATIC

For efficient de-icing protection, the correct amount of heat must be supplied where and when needed. If there is too little heat, the ice may not shed as required, perhaps causing large chunks of ice to shed or creating unbalanced rotors on a helicopter. If too much heat is supplied, there can be too much melting, resulting in undesirable amounts of runback ice. It has been found desirable to have the following characteristics in the cyclically heated shedding zones (reference 2-1):

a. A high specific heat input applied over a short period. This generally requires less total energy than a lower specific heat over a longer period of time. The high specific heat input reduces the convective heat losses from the exposed ice surface and conductive losses to the ice and structure, and to a lesser degree compensates for the uncertainty caused by the large variation in the bonding strength of ice. This requires that the deice system element-on-time (EOT) be optimized as a function of ambient or total air temperature to provide just the amount of heat necessary to melt the ice-to-surface bond layer. If the EOT is too short, de-icing will not occur. If the EOT is too long, the deicer will remain at an undesirably high temperature for a significant period of time, perhaps resulting in runback ice.
b. Immediate cessation of heating and rapid cooling of the surface after shedding occurs (this is to greatly reduce runback ice).
c. The heated area should be the minimum necessary to provide adequate coverage for all predicted icing encounters; heat not applied under the ice is dissipated to the airstream. Good insulation between the heater and the supporting structure is required to direct the heat outward to the exposed surface.
e. To produce clean shedding and to avoid runback icing, the proper distribution of heat is required. It is desirable that melting of the ice bond should occur uniformly over the surface; this may require some chordwise gradient to the heat input.
f. The spanwise and chordwise parting strips must prevent any bridging from one shedding zone to another. Even a small strip of anchorage may delay ice from shedding until dangerously large pieces have formed.
g. The cycle "off time" should be controlled to permit adequate ice accretion for the best shedding characteristics.

The "off time" will depend upon the thermal capacity of the shedding zone and the rate at which the surface cools to 32°F (0°C). It will also depend upon the icing rate so that the ice thickness accumulated is the best for shedding when de-icing occurs. The "off time" may be tailored to the maximum ice thickness allowed for the application. This may be as short as one minute for rotorcraft and as long as three to four minutes for fixed wing aircraft.

System requirements for de-icing must be considered in two parts: First, the parting strips and dividing strips, and second, the cycled shedding area. The heat input to the strips must provide at least running wet anti-icing to prevent ice from bridging. For electro-thermal de-icing, bridging means that an arch of ice remains over the heating element. The ice arch will be attached to a cool surface and the dividing strips can serve as attachment points if not heated. An arch can also form a bridge over the parting strip and be anchored on adjacent shedding zones. Control of the temperature of the strips is desirable both for economy and to prevent overheating. Severe overheating may cause a burnout of the heating elements. Dividing strip widths should be sufficient

to accommodate the change in stagnation point with changes in angle of attack. A parting strip width of from 1 to 1.25 inch (25 to 32 mm) should be adequate for most installations.

Aircraft wings with about 30° or more sweep-back will normally use only chordwise parting strips since aerodynamic forces will remove the ice from the shedding zones without the aid of stagnation line parting strips.

Shedding zone power requirements are difficult to determine. Heat transfer rates, aerodynamic forces, and bonding characteristics of the ice (to the surface material) are some of the factors affecting the power requirements. Heat-on times less than 10 seconds require heaters capable of withstanding very high power densities and a large number of cycled areas. Rotorcraft, with their smaller chord airfoils, have used element-on-times approaching one second for warm temperatures. Heat-on times longer than 40 seconds would not result in an appreciable reduction in power density. Shedding zone power density requirements are not appreciably affected by variations in heat-off times. To achieve an even melting of ice throughout the shedding zone it is desirable to have a uniform power distribution in the region of highest water collection, and a decreasing power requirement downstream.

Over-heat protection devices may be required for electrical de-icing systems. Inadvertent actuation of the electrical de-icing system during a period where little cooling takes place could cause burnout of the high wattage density heating pads. The over-heat protection device should be temperature sensing so that the systems may be used during ground operation to remove accumulated ice and during takeoff to prevent excessive ice formation. However, the system may not be very effective in ground operation due to lack of airflow to assist shedding.

An electro-thermal de-icing system may be installed on any existing aircraft by attaching flexible wrap-around heating pads to the structure with an adhesive. The heating pads are fairly thin and may not appreciably affect the aircraft performance. Another method of providing electro-thermal ice protection is to mold the heating elements into the leading edge structure. No aerodynamic performance loss is experienced by this construction method.

The electrical wiring supplying power for the heating pads requires little space and can normally be routed through existing passages within the structure. Generating equipment may be either engine-driven or auxiliary power source driven. The required power generator capacity will depend upon several factors: the efficiency of the heating pads, the wattage of the shedding sections, and the number of pads required to be on at any one time. Shedding section cycling should be scheduled to maintain an even power load upon the generating equipment.

In higher speed aircraft, the rubber heating pads are subject to rain erosion and hail damage. A metal overshoe (erosion shield) (figure 2-2) will protect the pads and help to even out the heat distribution and prevent surface cold spots. Detracting from the overshoe installation is the loss in heater efficiency and the added weight to the aircraft. The electro-thermal system for rotorcraft will generally be constructed in a composite, rather than in a rubber boot, but it will still require an erosion shield. Higher density de-icers are also often made from composites due to higher service temperature requirements.

2.3.1 Airfoil and Leading Edge Devices

2.3.1.1 Airfoils

The use of electro-thermal boot systems for all wing and empennage surfaces is usually not feasible because of the amount of power required and the associated weights of the power generating equipment. Wing surfaces provided with electro-thermal systems are usually limited to those requiring special consideration such as surfaces forward of engine inlets. These applications are generally anti-icing, running wet systems. De-icing is used for many empennage surfaces.

The heater elements are molded into the boot assemblies which are bonded to the wing leading edge. The boot is usually divided into a number of independent heating elements. Each heating element is protected by a circuit breaker and a current sensor. The temperature of the boot is controlled by a thermal control switch attached to the leading edge wing skin. The thermal control switch senses the skin temperature and opens or closes the power circuit at pre-set skin temperatures. See figure 2-3 for a typical wiring diagram. This system is activated continuously when in icing conditions. Also, the Temperature may be controlled via an electronic RTD controller, proportional control or ON/OFF.

2.3.1.2 Leading Edge Devices

The preceding description of an electro-thermal ice protection system for a wing assumed a fixed leading edge. Wings with high lift leading edge devices will have the same ice protection requirements but methods of satisfying these requirements will differ and will be more complex.

Leading edge devices may consist of slats, slots, or flaps. A wing leading edge slat configuration is shown in figure 2-4. A slot is similar to an extended slat but is a fixed geometry, rather than being retractable for cruising flight. Leading edge flaps are hinged plates which are nested against the wing for cruise and played forward for low speed, high lift conditions: See figure 2-5. The fixed wing behind the slat may or may not require protection, depending on geometry, ice accretion characteristics, and time in icing. For each new aircraft, a study should be made of the need for heating this area.

Electro-thermal de-icing may be applied to leading-edge slats. Difficulty in retraction of the slats may be experienced if ice accretes on the fixed leading edge or if residual ice or runback ice occurs behind the flap.

Leading edge "Krueger" flaps may or may not be protected depending upon the effects of ice accretion on performance. Figure 2-5 illustrates an unprotected leading edge flap as might be installed on a transport aircraft. Ice protection requirements for the wing leading edge will remain the same but because of the leading edge flap installation there will be a reduction in the heated area on the lower surface. Water may runback from the heated leading edge and freeze behind the flap if the leading edge anti-icing system is not fully evaporative. Also, there will be a small amount of direct impingement. Evaluation of the need for heating the flaps must be made by either an aerodynamic analysis or by flight test.

Ice accretion can occur when the flap is extended during takeoff and approach. Figure 2-6 illustrates the amount of ice that may accumulate on extended flaps during a 30 minute hold in icing. However, no appreciable ice usually accumulates during takeoff, as flaps are retracted about 1-1/2 minutes after brake release. If the flap is unprotected, flight tests should be conducted to determine the effects of this ice accretion. One flight test method which may be employed is to simulate the critical predicted ice shapes with wood or plastic and attach them to the flap. The airplane performance then may be evaluated in clear air.

It should be noted that the extension of either leading or trailing devices changes the air flow around the wing, including sizeable changes in stagnation line location. This will alter the droplet impingement and ice accretion patterns. Trailing edge flaps sometimes accrete ice; This can generally be determined only from flight tests.

De-icing requirements would be similar to the requirements of a fixed leading edge. Shedding characteristics will probably be different because of the airfoil shape and the aerodynamic forces the ice will encounter. An icing tunnel test program may be required due to the difficulty in predicting shedding forces and impingement.

2.3.2 Windshields

Anti-icing protection is usually provided for the forward-facing windshield panels on both military and commercial aircraft that are required to operate in all-weather conditions. The most widely used system is electrical anti-icing whereby electric current is passed through a transparent conductive film or resistance that is part of the laminated windshield. The heat from the anti-icing film or resistance wire also prevents internal fogging for most configurations. Electrical heat may also be used to maintain the windshield interlayers (of a glass/plastic laminated windshield) at or near the optimum temperature for resistance to bird strikes (birdproofing).

Figure 2-7 shows a typical windshield construction. The thickness of the conductive film or the wire size can be varied to accommodate variation in heating requirements or to heat irregular shapes (reference 2-2 and 2-3).

The windshield arrangement of a typical multi-engine transport is shown in figure 2-8. Ice protection is needed for the forward facing windshields (main and center), but not for the side and aft windows which are at minimum angle to the airstream and probably do not collect ice. An alternate arrangement often used deletes the center windshield and increases the size of the main windshields. The center windshield for rotorcraft was considered at one time not to require anti-icing, but operational experience showed that this section should also be protected. In either case, the conductive anti-icing film or wire grid is applied to the inside of the outer ply of glass, as shown in figures 2-9a-c. The film usually covers only a roughly rectangular area of the windshield, as non-uniformity of heating and actual "hot spots" become a serious problem with the high power density heating needed for anti-icing (3 to 4 watts/sq. in. for glass or 5.25 watts/sq. in. for plexiglass). The exterior ply is usually limited to 0.18 inch (4.6 mm) thickness if the surface is glass, 0.06 inch (1.5

power than de-icing systems but insure a minimum of engine ice ingestion, provided that runback icing is avoided.

2.3.3.2 Power and Cycling

Numerous power and timing combinations are available that can provide satisfactory ice protection. Alternating current obtained directly from aircraft alternators is typically routed through one or more electrical heating elements. Two fundamental parameters of electric power application are power density and duration. Typical power density ranges used in present day aircraft de-icing applications are from 10 to 30 watts per square inch (1.5 to 4.7 watts per square centimeter). Two different methods of applying power to the de-icing device are (1) holding power density constant and varying the duration - with the duration determined by ambient condition sensing - or (2) holding the time base constant and varying the power density (by varying the voltage or changing the total system resistance) - with power density determined by the ambient condition sensing. Regardless of which method is used, the success of a de-icing system in an inlet is heavily dependent on rapid heat-up and rapid cool-down. The desire to have rapid heat-up and cool-down grows out of the need to minimize the melting of ice and hence the forming of water during the de-icing cycle. By forcing the temperature of the surface to rise quickly, the ice will not gradually absorb heat, but instead, the frozen ice/surface interface will separate, allowing the ice particle to be swept away by the airflow, with the departing ice taking the small amount of melted water along with it. The more rapidly the surface cools to below freezing the sooner the reformation of ice will commence, preventing the runback of a significant amount of liquid water that could freeze onto unprotected areas. The potential problems of runback icing are dealt with in Section 2.10. A typical heating strategy to help alleviate the hazardous build-up of runback ice is to divide the deiced surface into several streamwise zones and apply power to each zone separately. By alternating the on-cycle from zone to zone and going from "front to back," runback water can be shed as ice or "chased back" to non-hazardous areas.

2.3.3.3 Materials

Electro-thermal de-icing heater elements are typically fabricated in one of several designs. The element may be built as an add-on pad type of device or it may be built in as an integral layer within the actual aerodynamic surface. If the heater is the bond-on pad type, it may be applied to either the front (air wetted) surface or on the backside of the aerodynamic surface, out of the air stream. If the heater is mounted on the external surface, consideration must be made for resistance to impact from foreign objects and from lightning strikes. Effects of changes in heat transfer characteristics should be accounted for when the decision is made to install a heater system externally, internally, or integrally. The installation location and consideration for impact resistance will usually have a bearing on whether the heater element is embedded in a flexible or a rigid material. In either case, the material must be electrically non-conductive and completely encase the heater element to insulate

the conductive element electrically. Fo. embedded heaters, an overtemperature sensor may be placed behind the heater to provide thermostat control and avoid heater damage by overheating.

2.3.4 Turbofan Components

Ice protection for turbofan engine components is commonly provided by hot air methods (Section III.5.3.4). However, electrothermal de-icing or anti-icing may be easily applied to inlet guide vanes.

2.3.5 Propellers and Spinners

Propeller and spinner (nose cone) ice protection is employed for three reasons: (1) leading edge ice formations may cause a propeller efficiency loss, (2) unsymmetrical shedding of ice may result in propeller unbalance, and (3) large pieces of ice shed from the propeller or spinner may be ingested by the engine on turboprop aircraft.

Electro-thermal de-icing may be installed on the external surface of a propeller without appreciably affecting its performance. Coverage should be determined by analytical or experimental means; it may be approximately 15 percent of chord on the suction surface and approximately 30 percent on the pressure surface. The maximum propeller radius at which appreciable ice forms is related to the increase in aerodynamic heating with increasing radius and also to centrifugal forces and rpm. Protection is sometimes extended no further than about 30 percent radius; however, in some instances, ice may accrete farther outboard with very detrimental effects on propeller performance since the propeller is most heavily loaded in the tip region. Careful selection of the protection coverage extremes is warranted in these cases. The power to the heating pads is supplied through slip rings at the propeller hub. Installation of the heating element on the interior of the propeller would protect the heating elements from damage, but the loss of efficiency and the difficulty of repair make the external system more feasible. An example of the construction of an external electric heater for a propeller is shown in figure 2-11. The outer ply of the propeller heater must be of a material which is rain erosion resistant and not too vulnerable to hail damage.

Propeller thermal de-icing requirements will vary depending upon propeller diameter, rotational speed, and aircraft forward velocity. The watt density and coverage are usually determined by the boot manufacturer. For composite blades, consideration must be given to runback due to the slower cooldown rate of the composite as compared to aluminum. This is usually accomplished by using a shorter ON TIME at warmer temperatures.

Shedding of ice from propellers should be equal from each blade of a propeller to prevent unbalance. Timing sequences are usually established so that only one propeller is heated at a time. Also, the frequency of shedding should be scheduled to prevent large pieces of ice from forming, as they may cause damage to the aircraft when they are shed. Fuselage skin damage due to ice shed in the propeller plane should be considered. In some cases, a protective shield may be desirable.

Ice protection for propellers and, to a limited extent, spinners of turboprop aircraft is commonly provided by the electro-thermal method. A typical system of this type is illustrated in figure 2-12 and is described in detail in reference 2-1. In this particular system, continuous (running wet) heating is provided for the forward portion of the spinner; cyclic heating is provided for the aft

Static ports located on the fuselage (if suitably far from the nose of the aircraft) usually have not needed protection. In any case, where ice protection is provided for small components, the effect of a single failure should be considered. In the case of pitot tubes, for example, if the pilot's pitot tube fails, the copilot's may be used.

2.3.8 Radomes and Antennas

2.3.8.1 Radomes

Electrical systems are not applicable to the ice protection of radomes because of the interference from the heating element (reference 2-5).

2.3.8.2 Antennas

Ice protection for antennas, if provided, can be of the electro-thermal type. General practice has been not to provide such protection, but large antennas such as those on the U. S. Army EH-60A helicopter are protected to avoid possible breakage due to large ice buildups. In some cases, flexible antennas improve the self-shedding of ice.

2.3.9 Miscellaneous Intakes and Vents

General practice has been to provide no protection for flush inlets and vents. Although detailed studies must be made for each new application, it is usually found that the flush inlets will not close completely in icing, therefore, protection is not provided. Static ports located on the fuselage (if suitably far from the nose of the aircraft) usually have not been protected. Inlets for air conditioning systems may or may not be protected depending on whether ice formed on the inlet would shed into the heat exchangers or turbo-compressors and cause damage. In any case, where ice protection is provided, it will probably be of the electro-thermal type.

2.3.10 Other

Other small components of an aircraft that accumulate ice if not protected are wing fences, stabilizer mass balance "horns," tip tanks, and wheel covers. The need for ice protection for these components can be assessed by answering the questions listed in Section 2.3.7. For many of these miscellaneous components, electro-thermal is the only practical ice protection method.

III.2.4 WEIGHT AND POWER REQUIREMENTS

The effect of an installed electro-thermal ice protection system on the weight and power requirements of some typical aircraft is presented in Table 2-1. From these data, the actual aircraft performance penalties can be predicted. These numbers are representative of what would be required; however, a more detailed analysis would be necessary to determine them more precisely for any particular airplane (reference 2-1).

III.2.5 ACTUATION, REGULATION, AND CONTROL

Automatic initial actuation of electro-thermal systems is discouraged for several reasons which generally apply to any type of ice protection system. First, automatic actuation decreases the reliability of the system. It could fail to be turned on, or be turned on when not required, reducing heating element life and wasting energy. Second, the pilot needs to determine if the system is functioning properly. His attention may not be drawn to a system that automatically actuates. Third, the complexity of the system is greater. If a manual backup is required, then an additional annunciator is required to show the primary system failure. More components require more frequent replacement and higher maintenance cost.

Having emphasized the dangers of automatic initiation, it must be stated that once activated, automatic regulation and control are very valuable in reducing crew workload. Annunciator lights or a CRT display should then be used to inform the pilot of system status or faults.

III.2.6 OPERATIONAL USE

2.6.1 Anti-Icing Systems

Anti-systems should be turned on as soon as icing conditions are encountered. If too much ice has accumulated, the system will be required to de-ice the surface and the watt density is usually too low to do this quickly.

These systems also need to be tested prior to flight into known icing conditions. In some cases, an annunciator light will indicate when the system is functioning properly. In other cases, the aircraft's ammeter or power meter are observed for reading changes when the system is turned on or off.

2.6.2 De-Icing Systems

De-icing systems do not have to be turned on until after ice has started to accumulate on the surface protected. The watt density is sufficiently high to remove accumulated ice. However, undue delay can result in the formation of ice over the parting strips, making shedding more difficult. Furthermore, if the system is not turned on before large amounts of ice accumulate, then some undesirable secondary effects may occur. For example, if operation of the propeller de-icing system is delayed too long, the ice thrown from the propeller would be thicker than usual and could cause excessive fuselage skin damage.

The de-icing systems need to be tested prior to flight into known icing conditions. This is done by turning on the system and observing the system ammeter or power meter to ensure the proper power is being provided. A periodic flicker should be observed when the system cycles from one area to another. The system would then be turned off until required. This conserves energy, extends the life of the heating element, and in some cases, keeps an element from burning out on the ground.

TABLE 2-1

Weight and Power Requirements for Electro-Thermal Ice Protection Systems For Three Typical Aircraft (Reference 2-1)

Aircraft	Aircraft Component	System Type	Weight Lb.	Power KW	Power HP
"A"	Windshield	Elec. Anti-icing	10	1.5	2.0
(Twin-	Propeller	Elec. De-icing	10	0.8	1.1
Engine		Generator (a)	15		
Recip.)	Total		35	2.3	3.1
	Wing and Tail	Elec. De-icing	40	6.8	9.1
	Windshield Elec.	Anti-icing	10	1.5	2.0
	Propeller	Elec. De-icing	10	De-iced with	
		Alternator (b)	20	wing and tail	
	Total		80	8.3	11.1
"B"	Windshield	Elec. Anti-icing	10	1.4	2.0
(Single-	Propeller	Elec. De-icing	5	0.6	0.8
Engine		Generator (a)	15		
Recip.)	Total		30	2.0	2.8
	Wing and Tail	Elec. De-icing	45	9.3	12.5
	Windshield	Elec. Anti-icing	10	1.4	1.9
	Propeller	Elec. De-icing	5	Deiced with	
		Alternator (b)	20	wing and tail	
	Total		80	10.7	14.4
"C"	Wing and Tail	Elec. De-icing	45	11.7	15.7
(Light	Windshield	Elec. Anti-icing	10	1.5	2.0
Twin Jet)	Engine Inlet	Elec. Anti-icing	10	3.8	11.8
		Alternator	20		
	Total		85	22.0	29.5

(a) 15 lb. added for increase in generating capacity (28-volt generator + transformer).

(b) 20 lb. added for increase in generating capacity (115-volt alternator substituted).

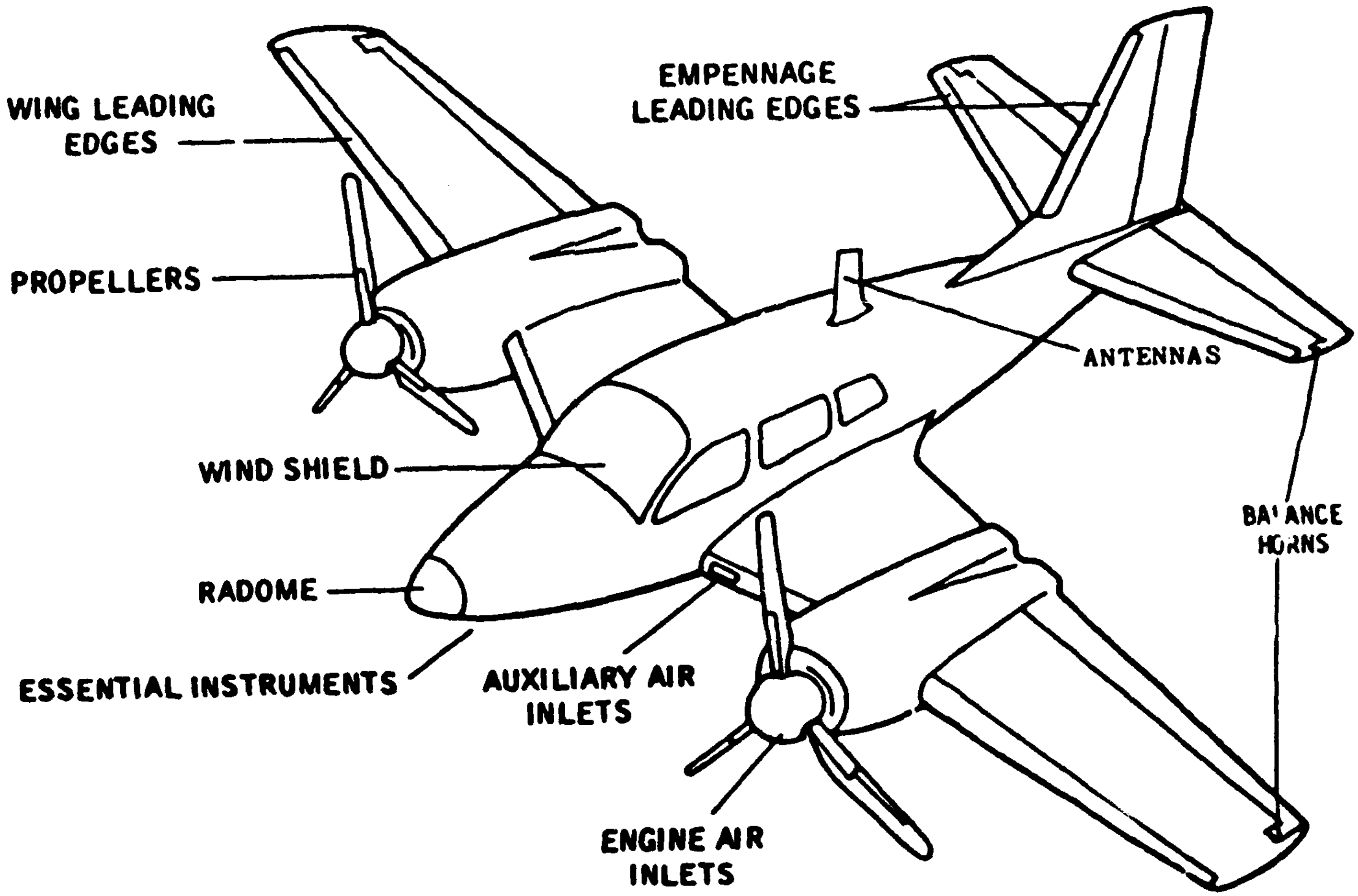


FIGURE 2-1. AREAS OF AIRFRAME THAT MAY REQUIRE ICE PROTECTION

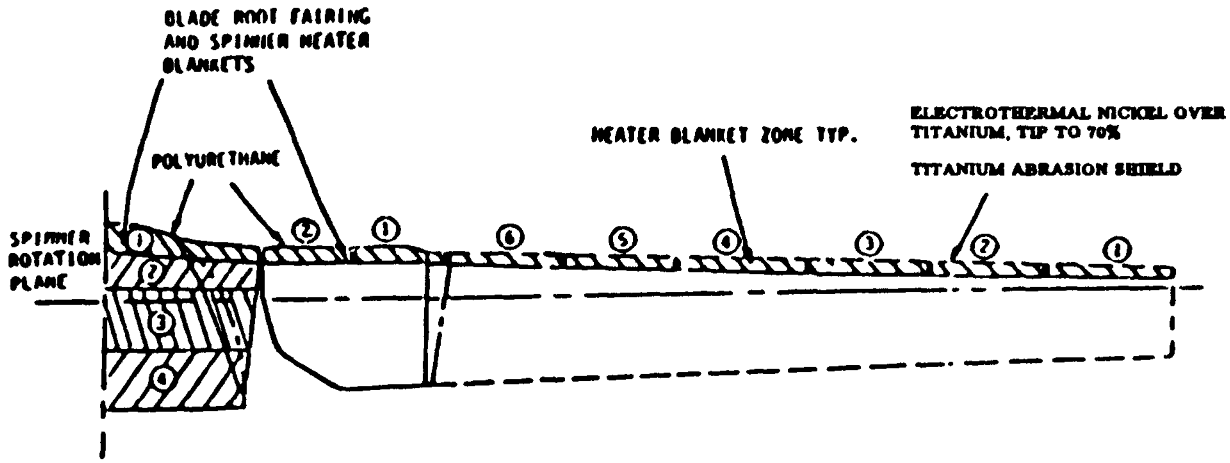


FIGURE 2-14. SPANWISE HEATER ELEMENT ARRANGEMENT

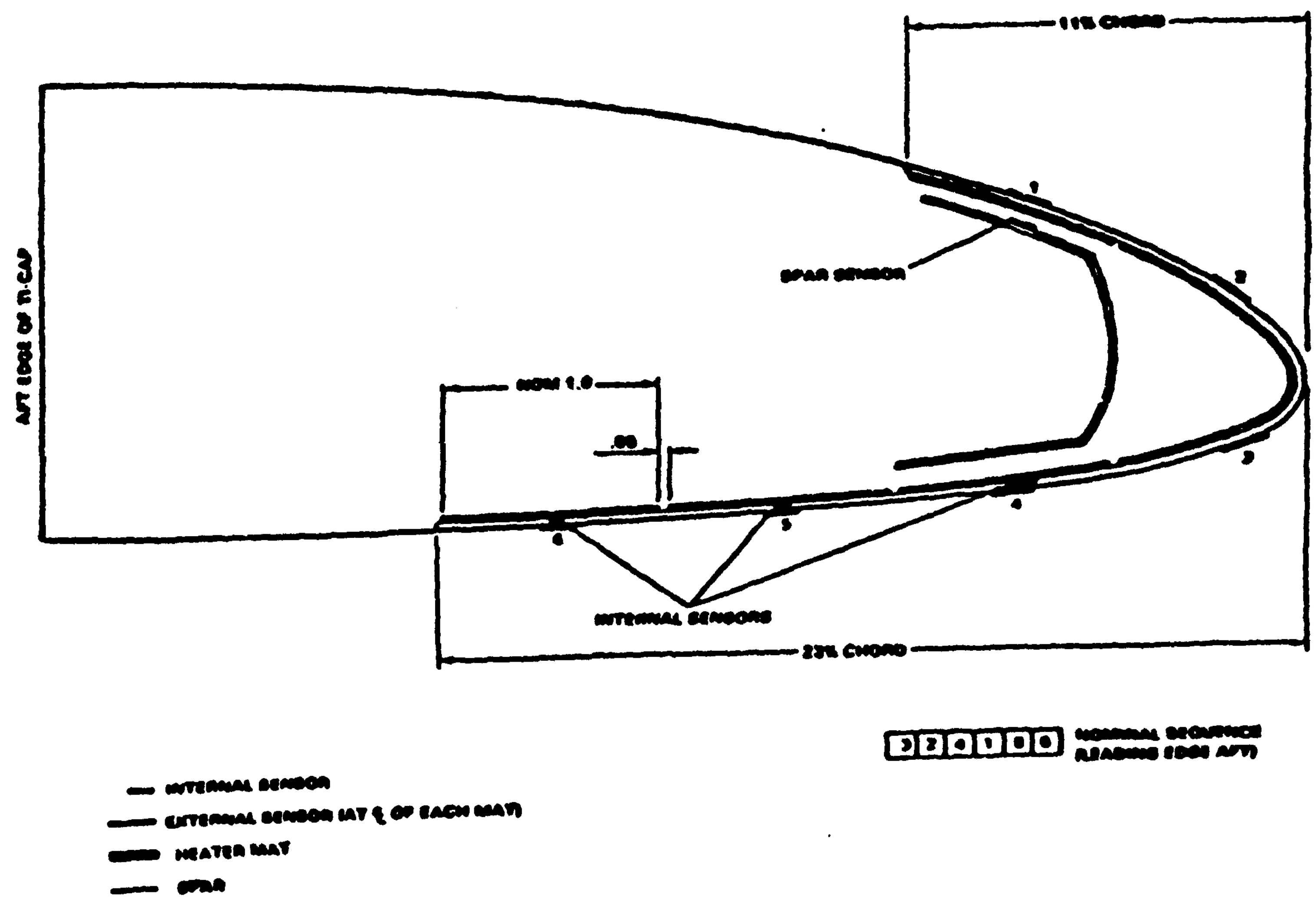


FIGURE 2-15. CHORDWISE HEATER ELEMENT ARRANGEMENT

CHAPTER III - ICE PROTECTION METHODS

CONTENTS

SECTION 4.0 ELECTRO-IMPULSE SYSTEMS

LIST OF FIGURES

III.4.0 ELECTRO-IMPULSE SYSTEMS

III.4.1 OPERATING CONCEPTS AND COMPONENTS

Electro-Impulse De-Icing (EIDI) is classified as a mechanical ice protection method. Ice is shattered, de-bonded, and expelled from a surface by a hammer-like blow delivered electro-dynamically. Removal of the ice shard is aided by turbulent airflow; thus, relatively low electrical energy is required.

Physically, the EIDI system consists of ribbon-wire coils rigidly supported inside the aircraft surface to be de-iced, but separated from the skin surface by a small air gap (figure 4-1). A high voltage (typically 800 to 1400 V) electric current is discharged through the coil (figure 4-2). The circuit must have low resistance and inductance to permit the discharge to be very rapid, typically less than one-half millisecond in duration. A strong electromagnetic field forms and collapses, inducing eddy currents in the aircraft skin. The eddy current and coil current fields are mutually repulsive, resulting in a toroidal-shaped pressure on the skin opposite the coil. The peak force on the skin is typically 400-500 pounds (1780-2220 Newtons) and is delivered so sharply as to produce a sound resembling a metal-on-metal blow. Actual surface deflection is small, generally less than 0.01 inches (0.25 mm), but acceleration is rapid.

The coil may be supported from a spar, a beam between ribs, or from the skin itself. In any case, the coil's mount or direct support should be non-metallic to avoid interaction with the coil's magnetic field. At a leading edge spanwise station there may be a single coil at the nose, a pair of coils at upper and lower skin positions near the nose, or even a single coil placed eccentrically on either upper or lower surfaces.

During EIDI systems operations a coil receives 2 or 3 successive pulses from the capacitor unit with pulses separated by the time required to recharge the capacitor, typically 2 to 4 seconds. The spanwise extent of wing leading edge that each coil (or coil pair) will de-ice depends largely on the structural properties of the leading edge, but it is nominally 18 inches (0.5 meters). The capacitor is then switched to another coil station, and then to another until it cycles around the aircraft. The time to complete the de-icing cycle must be less than the time for acceptable ice accretion for the protected surfaces.

The system consists of a power-and-sequencing box, often located in the fuselage, and a number of coils in the wing, empennage, and engine inlet lip surfaces which are connected to the power box. Figure 4-3 shows the system in its simplest form. The capacitor discharge occurs when a solid-state switch is remotely triggered to close the circuit. This high voltage, rapid response switch is a Silicon Controlled Rectifier (SCR) or "Thyristor". Gas tube thyristors ("thyratrons") are also available but have not been used in the USA for EIDI. The circuit often includes a "clamping" diode, as shown in figure 4-2, to prevent reverse charging of the capacitors.

The first nation to use EIDI was the USSR, which had a fully equipped aircraft in 1972 and has equipped several transport-sized airplanes since (reference 4-1). No operational data are available.

The EIDI system has had extensive testing in the NASA Lewis Research Center's Icing Research Tunnel (references 4-4 and 4-6) and limited flight testing in the USA by NASA and Cessna Aircraft Company (references 4-6 and 4-7). Other testing has been done by Boeing Commercial Airplane Company (including a flight test series in a B-757), Rohr Industries (icing tunnel tests of engine inlets), Douglas Aircraft Company (laboratory and icing tunnel tests), Wichita State University (Fatigue and Electromagnetic Interference Tests, reference 4-8) and Electroimpact Inc. (Electromagnetic Testing of Modular Low Voltage EIDI Systems, reference 4-9).

III.4.2 DESIGN GUIDANCE

4.2.1 Pulse Width Matching

The EIDI system requires a careful and rather sophisticated design (references 4-2, 4-3, 4-8, 4-10 and 4-11). The current pulse width in the coil resulting from the capacitor discharge must be properly matched to the skin electrical properties (reference 4-4) and to the leading edge structural dynamic response (reference 4-5). Failure to do this properly severely reduces the coil's ice expelling performance. When skin conductivity is too small, a higher conductivity metal disc is bonded to the inside of the skin opposite the coil. This disc is termed a "doubler" and should be slightly larger in diameter than the coil (figure 4-1). Copper or unalloyed aluminum are the common doubler materials. Doublers increase the skin stiffness locally but may distribute the impulse load more evenly. A structural dynamic analysis will provide guidance for the proper placement of the coil for maximum efficiency.

4.2.2 Power Supply and Sequencing

The power supply and sequencing may be packaged in a single box. The sequencing system can be confirmed for a single sequence around the aircraft or for continuous resequencing. Power supplies are available for common aircraft voltages and frequencies.

Installation of the power supply and control system in the aircraft should be done in a manner that minimizes the distance through which the high energy electrical pulse must travel. Ideally, the capacitors should be located in proximity to the coils, while the power supply, with its "trickle charge" to the capacitors, should be located near the aircraft electrical generator and away from the capacitors. Figure 4-4 shows a system schematic, and figure 4-5 shows a large airplane application. Each aircraft will require a trade-off study to determine the number of coils to be supplied by one capacitor. For large aircraft, the weight and electrical losses of the high current lines requires several capacitor sets. However devoting a capacitor to each coil pair may result in a costly, heavy system, so a compromise between those extremes is needed.

Redundancy can also be obtained by using multiple power boxes which are cross connected. This provides power to all coils at longer time intervals if one box fails. An alternate safety system, which increases cabling, is to supply alternate coils from different power sources. That is connect odd numbered coils to one power supply and even numbered coils to another. In case of one power box failure, only limited amounts of ice will collect on the unprotected area, unless very stiff nose ribs separate the leading edge segments.

Note in figure 4-5 that the SCRs ("thyristors") are placed on the capacitor side of the coil to avoid having high voltage on coils not being fired.

4.2.3 Coil Design and Installation

Coil placement and mounting methods are critical. Coil mounts must be quite rigid to avoid energy losses due to mount flexing. Mounts are generally made of composite material. Typical coils are wound from copper ribbons and are about 2 inches (50 mm) in diameter, 0.12 to 0.20 inches (3 to 5 mm) thick, and contoured to match the leading edge skin inner shape. A layer of insulating enamel and a thin layer of fiberglass cloth are usually bonded over the coil. The air gap between the leading edge inner surface or doubler and the coil must be sufficient so that the vibrating skin will not strike the coil.

Special attention must be paid to the attachment methods for the doubler since very high loads will be passing through the doubler into the leading edge skin. If attachment of the doubler is done with mechanical fasteners, it should be done in a manner such that no part of the fastener will be drawn into the engine in event of fastener failure. However, adhesives are available for bonding of doublers, and so mechanical fasteners are not required for most installations.

III.4.3 USAGES AND SPECIAL REQUIREMENTS

4.3.1 Airfoil and Leading Edges

Coils are generally wired so that two spanwise positions are in series to reduce the number of cables and thyristors and to achieve better structural response. The two coil positions wired in series can be adjacent span stations or alternate positions. The thyristors (or "SCR"s") are usually located near the surface to be deiced to permit use of a common supply cable for several coils (figure 4-3).

Installation is most satisfactorily accomplished as original equipment at the factory. Retrofitting can be accomplished, but may require structural modification to the leading edge to suit coil placement and spacing. The small radius of curvature of empennage leading edges on small airplanes often precludes the use of a single nose coil, requiring two smaller opposing coils at the upper and lower sides of the leading edge.

4.3.2 Windshields

EIDI is not recommended for windshield de-icing.

4.3.3 Engine Inlet Lips and Components

Icing Wind Tunnel tests have been conducted on an EIDI system installed in a nacelle (figure 4-6) from a business jet aircraft (references 4-3 and 4-12). These tests show EIDI to be a viable method for aircraft engine inlet lip ice protection. Since EIDI expels ice particles, some of which are ingested by the engine, ice pieces were collected in a net for examination. From these observations and high speed photographic studies, the general rule was proposed that the effective diameter of particles expelled will be no larger than three times the maximum thickness of the ice layer. For these specific observations ice particle thickness was no larger than 1/8 inch and it was concluded that these particles could be ingested by the engine used in this application without damage. In addition, this type engine ice ingestion must not cause an engine flameout. This requirement may call for more frequent impulses than for a wing or empennage protection system. No difficulty was experienced in de-icing due to added stiffness inherent in the inlet lip compound curvature. For inlets tested, a single coil on the inlet lip inner portion was better than either a nose coil placement or an inside-outside pair of coils. Spanwise spacing of the coils was about the same as for small airplanes, nominally about 18 inches (0.5 m). This system offers a significant reduction in the energy needed for ice protection when compared to hot air (bleed air) anti-icing systems. The potential applications for EIDI in engine inlets are numerous and encompass large-diameter, high bypass ratio turbofans, small-diameter business jet engine inlets, and circular or noncircular turboprop engine inlets. Considerations other than inlet type or shape will be the determining factor in the selection of EIDI as the best ice protection system. Initially, a determination should be made of the ice ingestion capability of the engine. EIDI can be designed to remove ice of a specified thickness, with larger thicknesses becoming progressively easier to remove. If the engine can handle short periods of cyclic ice ingestion of a specified size without undue compressor or fan blade erosion in the long term, EIDI can be safely used. The relatively small pieces of ice are a product of the removal mechanism that shatters the ice build-up. The ice will not shed in large continuous pieces if the system is properly designed and operated.

4.3.4 Turbofan Components

EIDI has not been applied to engine components such as inlet guide vanes because of their small size.

4.3.5 Propellers, Spinners, and Nose Cones

EIDI is not considered applicable to propellers because of their small blade cross section and rigid structure in the small radius portion which has the greatest tendency to accrete ice.

Spinners and nose cones can be deiced by electro-impulse. Coils can be supported on mounts fastened to nearby structure or can be skin-mounted. Nonrotating nose cones can be wired in the same manner as wing leading edge coils. Rotating cones or spinners introduce the complexity of commutator rings to transmit electrical power. It is generally poor practice to transmit the EIDI pulse

across commutator rings due to the high currents and voltages involved. This suggests placing the capacitors in the spinner, and transmitting low voltage recharge current across the commutator ring. A separate charging and firing circuit is then required for each spinner. This complexity may limit EIDI use for spinners.

4.3.6 Helicopter Rotors and Hubs

Application of EIDI to rotorcraft rotors is still in the development stage at the present time (1992). Retrofit is usually not possible because no leading edge cavity exists in which to place the coils and cable runs. Because of the critical balance and aeroelastic requirements, the EIDI equipment should he designed into the blade at the factory. An unbonded metal leading edge will be required on a rotorcraft blade. This is usually the abrasion shield which is fitted tightly over the leading edge substructure and bonded only at the downstream edges. The coils may be recessed into the leading edge composite material with a gap between the coil face and the abrasion shield. If the abrasion shield has insufficient conductivity, a doubler of higher conductivity material may be bonded to it opposite the coil location.

Present design planning to use EIDI in rotorcraft locates the power and sequencing boxes in the rotating hub column, with a commutator ring bringing the low voltage current to a transformer, and a rectifier in the hub for continuous recharging of capacitors. Care must be taken to de-ice opposing blades symmetrically.

The damping effect of the rotor sub-structure on the metal surface makes it necessary to have coils at closer spacing intervals than for the fixed wing hollow structures. The small geometry usually dictates the use of a coil at the lower side rather than at the nose of the leading edge. Once the power and sequencing unit is provided in the rotor hub, the addition of coils to deice the hub's external surface is as easily accomplished as for a wing leading edge.

4.3.7 Flight Sensors

EIDI is not applicable to flight sensing instruments.

4.3.8 Radomes and Antennas

Aircraft radome and antenna de-icing have not been accomplished with electro-impulse. Before using EIDI coils for such de-icing, the possibility of electromagnetic interference with the transmitter/receiver should be evaluated.

4.3.9 Miscellaneous Intakes and Vents

The minimum diameter of impulse coils is approximately 2.5 inches (66 mm) and only components with structural voids large enough to permit installation are possible candidates for EIDI protection.

4.3.10 Other

EIDI coils can be varied in configuration and size for installation in structures whose remote locations or complex shapes make them difficult to de-ice by thermal or pneumatic boot systems. This is particularly true of surfaces which contribute drag but do not reduce lift when iced. Struts which are aluminum extrusions used for wing braces on small airplanes are easily de-iced because of their structural dynamics (reference 4-4). Other candidates for EIDI usage are engine pylons and wheel covers.

III.4.4 WEIGHT AND POWER REQUIREMENTS

Estimates of weight and power required for EIDI are tentative at this early stage of development. The data presented below are, nevertheless, considered by the system developers to be conservative:

Aircraft	Power*	Weight**
6-place, propeller driven	400 watts	60 lbs***
150 passenger turbofan transport		
no redundancy:	2 kilowatts	250 lbs
full redundancy:	2 kilowatts	400 lbs
250 passenger transport		
no redundancy:	3 kilowatts	350 lbs
full redundancy:	3 kilowatts	500 lbs

* Based on 3 minute cycle for all surfaces.
** For wings and empennage surfaces; for engine inlets, increase by 25%.
*** Also includes wing struts for small airplanes.

III.4.5 ACTUATION, REGULATION, AND CONTROL

For critical surfaces not visible to the pilot, such as inboard wing areas or empennage, use of some type of ice detection sensor may be advisable. EIDI activation can be either manual, with appropriate cockpit display, or automatic.

III.4.6 OPERATIONAL USE

A simple test of small airplane systems can be performed on the ground by placing one's hand on the leading edge skin as each coil fires. Audible differences are evident for coils whose mounting has failed or whose circuit contains an electrical fault. An oscilloscope view of the current from the capacitor box may reveal changes in EIDI system physical geometry or electrical circuit faults. The more sophisticated units may have test circuitry installed in the aircraft for inflight system checkout.

III.4.7 MAINTENANCE, INSPECTION, AND RELIABILITY

Lack of sufficient operational experience at this time does not permit assessment of maintenance requirements or reliability. The power and sequencing box must be accessible for inspection. Terminals should be available in the box for attaching test leads to each coil's set of wires. For units using electrolytic capacitors, degraded performance may result if the capacitors are operated at 14 to -22 °F (-10 to -30 °C) temperatures (reference 4-14) and damage to the capacitors may result at temperatures below -40 degrees F (-40 degrees C). Inspection tests should evaluate capacitance after cold exposure. For many uses, the more costly metalized capacitors are required; these are not damaged by low temperature.

III.4.8 PENALTIES

See limitations below.

III.4.9 ADVANTAGES AND LIMITATIONS

Advantages of the EIDI System are:

a. Low power required. EIDI system power consumption is less than 1% of that required for hot air or electro-thermal anti-ice systems. Power requirements for an EIDI system may be about the same as for the landing lights for the same aircraft (see Section 4.4 above).

b. Reliable de-icing. Ice of all types is expelled, with only light residual ice remaining after the impulses. Ice thicknesses from 0.03 to 1.0 inch (8 to 26 mm) have been consistently shed (figure 4-7).

c. Non-intrusive in the airstream, hence no aerodynamic penalty.

d. Weight comparable to other deicing systems.

e. Low maintenance, theoretically, since there are no moving parts; however coils, capacitors, and diodes can fail.

f. No run-back re-freezing occurs.

g. Pilot skill and judgement required to operate the system are minimal in that no threshold ice-thickness is required for turn-on.

h. System can include self-test circuitry.

Limitations of the EIDI system are:

a. It is new and has limited use at this writing (1992).

b. It is not an anti-icing system, so some ice will be present over most of the aircraft leading edges during flight in icing.

c. Outside the aircraft the discharges may be quite loud, resembling a light gunshot. Inside small aircraft, the impulses are audible but may be almost imperceptible in a large transport category airplane.

d. Complex design requirements.

III.4.10 CONCERNS

Concerns not resolved because of lack of operational experience are:

a. Possible fatigue of skin, coil mounts, insulation, etc. Testing indicates that coil mountings must be well designed to avoid fatigue failure. Laboratory testing has been done with small airplane leading edges, both aluminum and composite, at low temperatures and for normal de-icing electrical engines. No fatigue damage was found after impulses equaling the number expected in a twenty-year aircraft lifetime. Similar laboratory tests for a transport aircraft slat exceeding 200,000 impulses showed no fatigue damage (reference 4-8). All of these used doublers with the skins.

b. Electromagnetic interference (EMI). The discharge of 1000 volts to create transient electromagnetic fields might he expected to cause undesirable signals in communication, control, or navigation equipment. However, both laboratory and flight tests have failed to detect appreciable interference for metal air foils (references 4-6 and 4-7). The reasons suggested for this are: (1) the aluminum wing box provides a good shield (note that this might not be true for a non-metallic leading edge); (2) the frequency of the pulse is below 3 khz, which is below that of current aircraft avionics systems; and (3) the pulse is a pure wave (or half wave) without the "overtones" of a spark. In flight tests, added equipment has been carried specifically to detect EMI; these included LORAN-C, digital readout systems, and a radar pod mounted on the wing. One of these had control wiring which shared space with the EIDI cables. However, care should be taken to check for EMI, especially if non-shielded leads are physically near the EIDI power cables.

c. Possible adverse effects of lightning strikes. Since the EIDI system is electrical, there is the possibility of its being disabled by a lightning strike. Sudden overload protection of critical components may be required. If the EIDI system is installed in an aircraft whose structure is largely of composite materials, the EIDI cables could become the primary electrical paths through the structure when it is struck by lightning. Additional lightning paths through the aircraft may be required.

d. Failure modes and their consequences have yet to be clearly defined. Compliance with the failure analyses as described in FAA AC 25.1309-1A (reference 4-13) is required. The failure modes and redundancy possibilities will be quite different, however, for small and large aircraft.

e. See reference 4-14.

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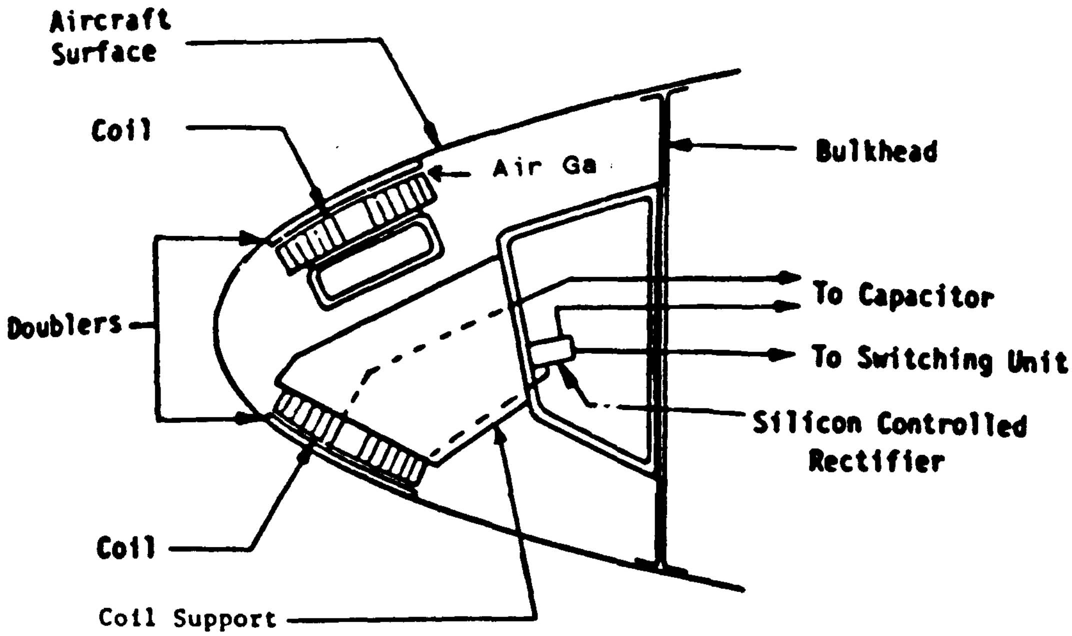


FIGURE 4-1. IMPULSE COILS IN A LEADING EDGE

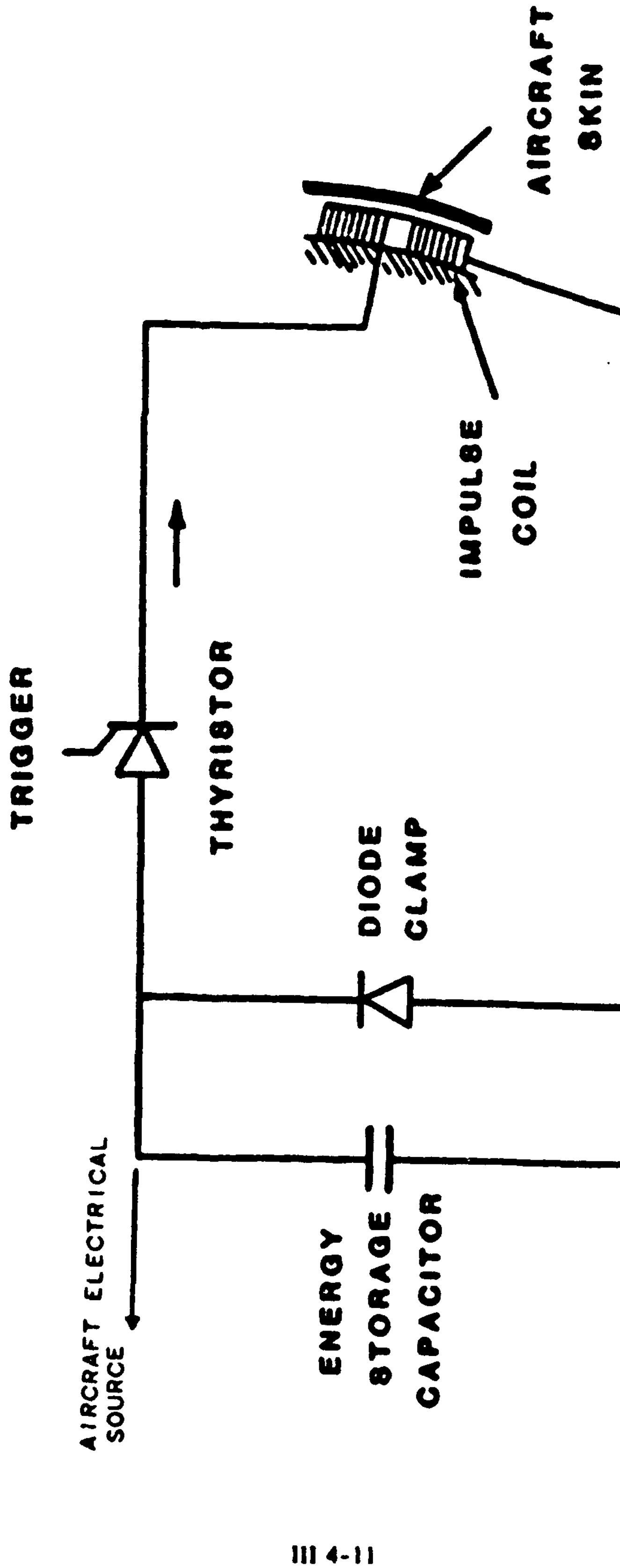


FIGURE 4-2. BASIC CIRCUIT

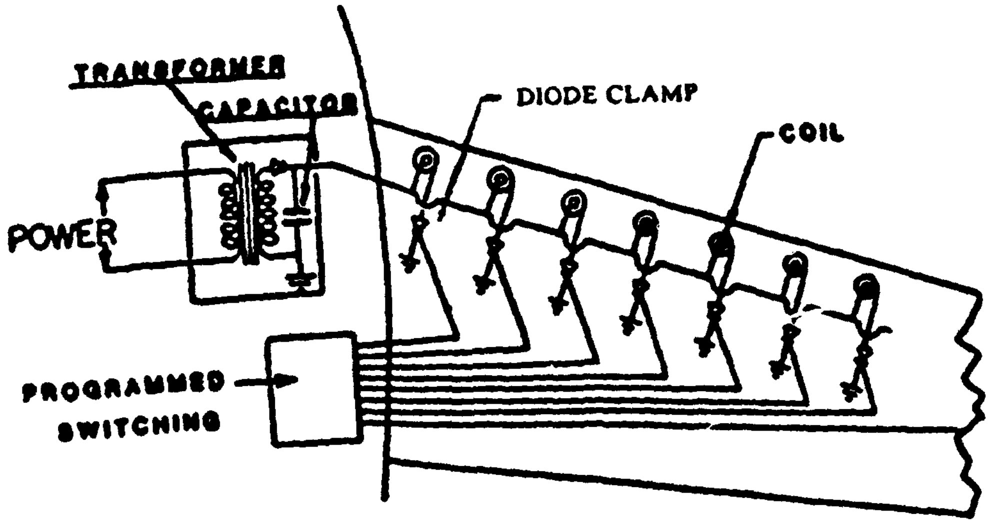
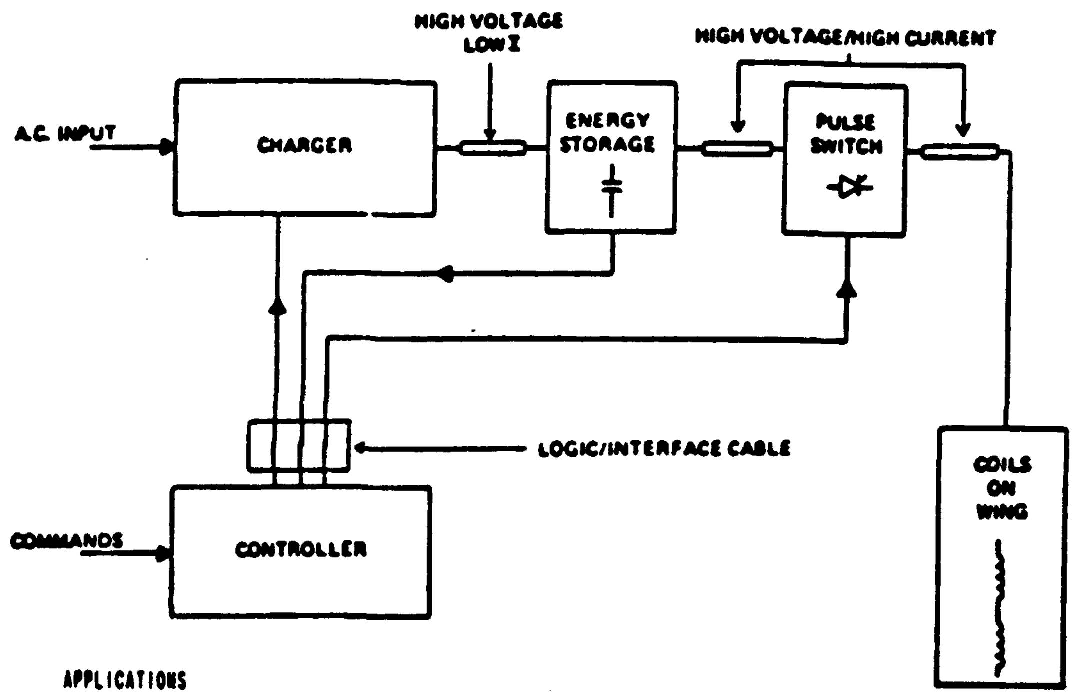


FIGURE 4-3. A BASIC ELECTRO-IMPULSE DE-ICING SYSTEM IN A WING

THE IMPULSE DE-ICING SYSTEM
FULL FEATURE SYSTEM



APPLICATIONS
- LARGE TRANSPORT
- HIGH TECHNOLOGY AIRCRAFT
- ENERGY EFFICIENT AIRCRAFT

FIGURE 4-4. EIDI SYSTEM SCHEMATIC

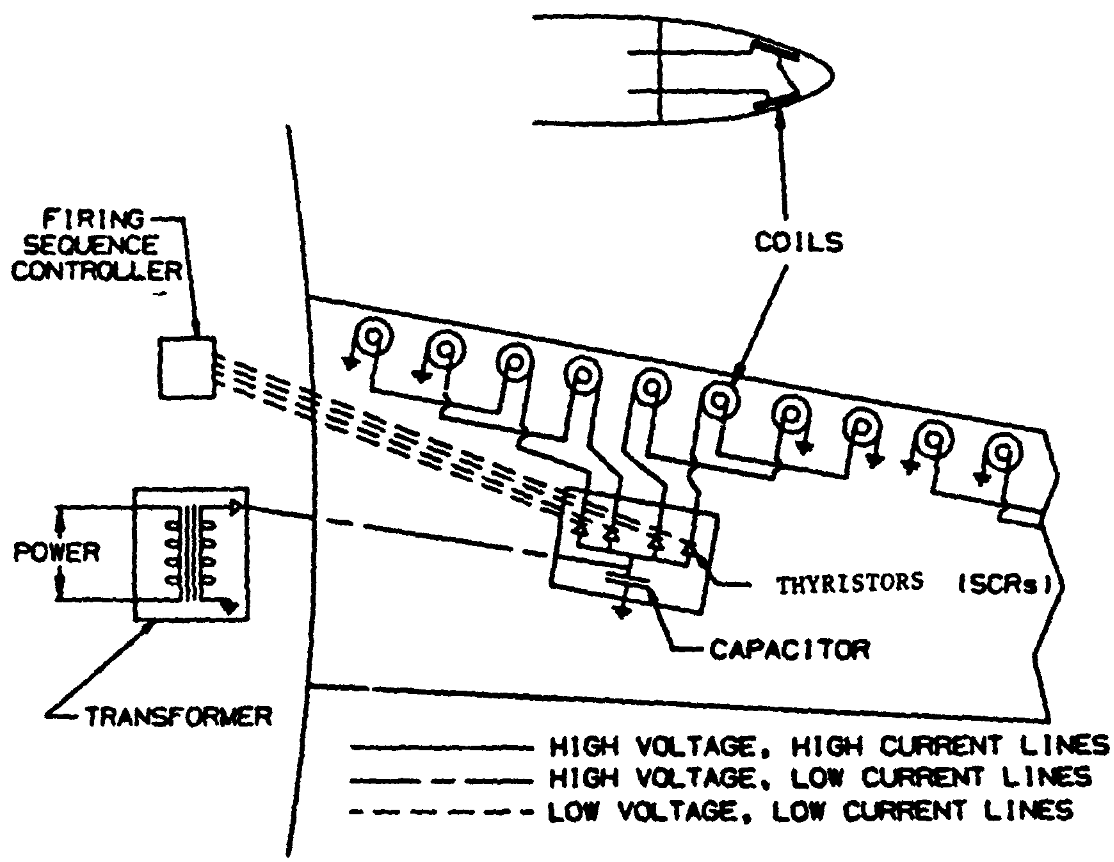
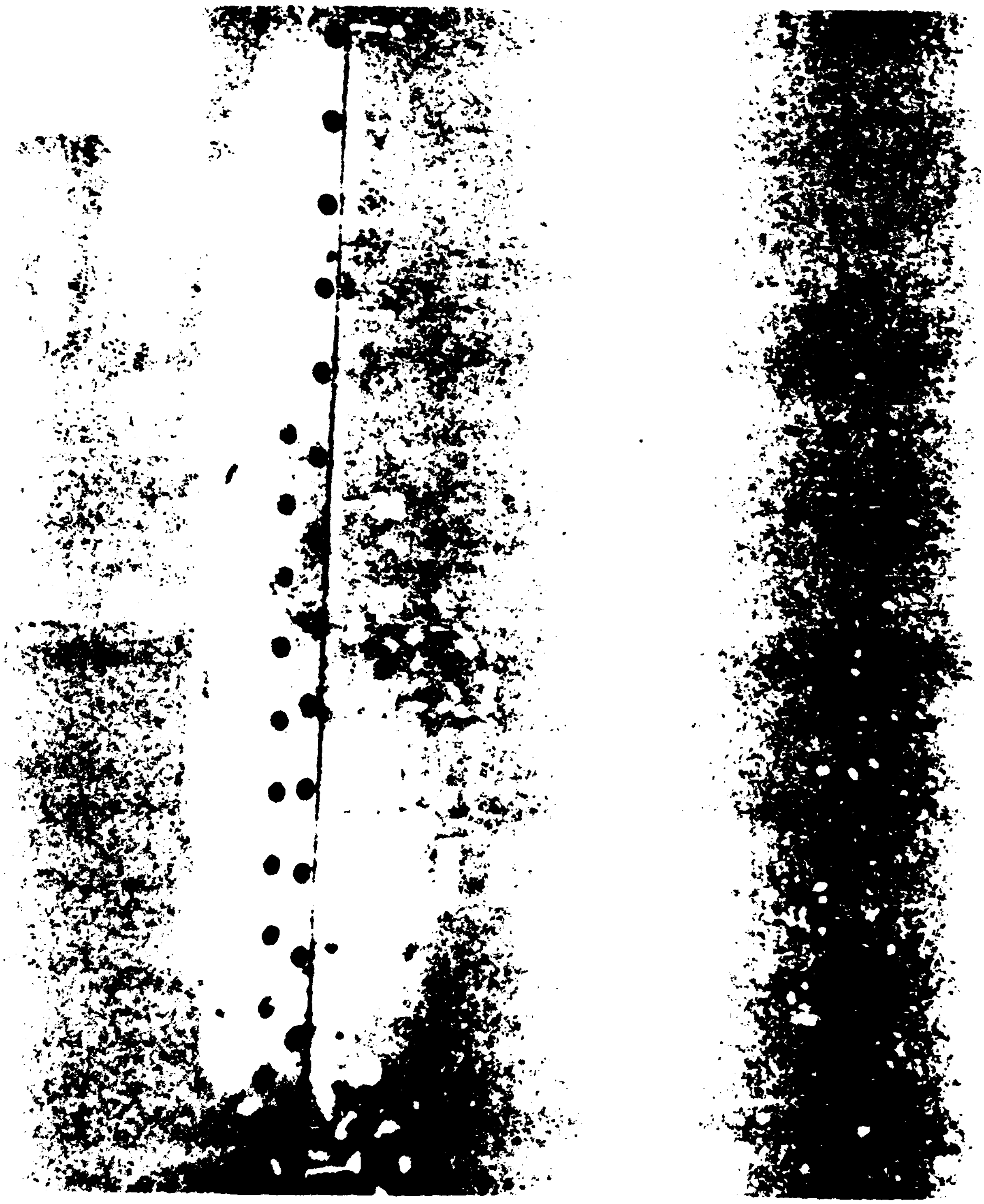


FIGURE 4-5. EIDI SYSTEM IN A LARGE AIRCRAFT WING

A. Accreted Ice on the Engine Inlet

B. Accreted Ice from the Engine Inlet Has Been Removed by EIDI

FIGURE 4-6. FALCON FANJET ENGINE INLET BEING DE-ICED BY EIDI



CHAPTER III

SECTION 4A.0

ELECTRO-EXPULSIVE DE-ICING SYSTEMS

CHAPTER III - ICE PROTECTION METHODS

CONTENTS

SECTION 4A.0 ELECTRO-EXPULSIVE DE-ICING SYSTEMS

LIST OF FIGURES

SYMBOLS AND ABBREVIATIONS

Symbol	Description
AC	Alternating Current
°C	Degrees Celsius
DC	Direct Current
EEDS	Electro-Expulsive De-icing System
EMI	Electromagnetic Interference
°F	Degrees Fahrenheit
FAA	Federal Aviation Administration
SCR	Silicon Controlled Rectifier
V	Volts
VAC	Volts Alternating Current
VDC	Volts Direct Current

GLOSSARY

dielectric - An insulator or non-conducting electrical medium.

elastomeric - Any substance having the properties of rubber.

electromagnetic interference - The field of influence produced around a conductor by the current flowing through it which contributes to a degradation in performance of an electronic receiver. Also called electrical noise, radio interference, and radio-frequency interference.

equivalent spherical diameter - The uniform diameter an ice shard would have after melting into a liquid water droplet.

G-force - A dimensionless descriptor relative to normal of the force acting upon an object due to gravity where two G's would infer a weight doubling due to twice the gravitational pull upon the object mass. Also used to describe acceleration or centrifugal reactive forces.

neoprene - Any of a group of synthetic rubbers. A non-conductor of electricity and superior to rubber in wear resistance.

polyurethane - A strong plastic resin that resists fire, weather and corrosion made in flexible or rigid materials. A non-conductor of electricity.

silicon controlled rectifier - A semiconductor device that functions as an electrically controlled switch for dc loads. Also known as a "thyristor" which is the solid state equivalent of a thyratron vacuum tube.

III.4A.0 ELECTRO-EXPULSIVE DE-ICING SYSTEMS

III.4A.1 OPERATING CONCEPTS AND COMPONENTS

Electro-Expulsive De-Icing Systems (EEDS) are classed as mechanical ice protection systems because accreted ice is expulsed from blanket-protected structures by a strong and rapid movement of the blanket outer weathering surface, which overlies parallel electro-expulsive conductors and a non-conducting elastomeric matrix. This impulse movement results from an electrical current being pulsed in opposite directions through closely-spaced parallel conductors or conducting layers imbedded in a non-conducting elastomeric matrix within the blanket (figure 4A-1). An electromagnetic force is thus created that acts to move the conductors or conducting layers apart. With the bottom set or segment of conductors stationary, and the top set or segment moveable (figure 4A-2), this separation force accelerates the top surface of the blanket outward so as to destroy the ice-blanket bond and inertially expel the surface ice into small pieces. Ice removal is accomplished by aerodynamic forces, centrifugal forces, and to a minor extent, gravity. In the most efficient EEDS designs, a minimum amount of momentum is added to the ice as a result of the expulsive force exceeding the strength of the ice-blanket bond. Actual displacement of the blanket surface is a function of ice thickness. Maximum displacement, on the order of 0.025 to 0.050 in., occurs when there is no ice and the elastomeric matrix is warm. The time for a blanket surface to return to a rest position is on the order of 0.001 seconds.

The electro-expulsive de-icer conductors are in a self-contained elastomeric medium (blanket) which is bonded onto or integrated into the leading edge surface and protected by an outer weathering surface. Other primary components (figure 4A-3) of an EEDS are some form of controller, energy charging and storage unit(s), energy switching and distribution unit(s), and a cockpit control panel. Miscellaneous components include high-current, low-inductance coaxial cabling, electrical interface wiring, connectors, and the appropriate circuit breakers and/or fuses. All of the EEDS operating equipment can be tailored to operate from either 28 VDC, 115 VAC single-phase or 115/200 VAC three-phase configurations.

III.4A.2 DESIGN GUIDANCE

4A.2.1 De-Icer Blanket

De-icer blankets differ somewhat between manufacturers but basically consist of an outer weathering surface, an electro-expulsive power circuit (conductors), and a dielectric (non-conducting) elastomeric support matrix. These materials are capable of operating in the temperature range of -67 to 250 °F (-55 to 121 °C). Blankets are smooth on both sides and nominally from 0.03 (at the leading edge) to 0.08 inches thick. The outer surface of the blanket may be metallic or elastomeric (polyurethane or neoprene), depending upon the application. The metallic surfaced blanket is more

difficult to install but has better erosion characteristics and fewer maintenance concerns. The elastomeric surfaced blanket is easier to install and more applicable to retrofit applications.

In addition to providing an aerodynamically smooth surface, the erosion layer also provides a continuous surface to transmit the in-plane horizontal pressure wave of an EEDS blanket. This pressure wave provides ice-shed forces across butted segment joints and also slightly beyond the edge of the outer most conductor.

Blanket materials can be quite simple. This allows material changes for specific reasons with only modest functional test and process verifications. This is advantageous since blanket materials need to be compatible with a number of substrates, adhesive systems, and external environmental conditions. For example, the outer weathering surface can be changed as material improves. This also permits a potentially simple maintenance philosophy. Should a segment be damaged or fail, the erosion layer is removed, a replacement segment installed, and a new top layer installed to complete the repair.

In addition to the type of blanket installation, consideration must be given to the design and layout of the individual de-icer zones or segments within the blanket. These may be configured in many shapes, and either chordwise or spanwise, depending upon the particular application. Although a single-segment blanket is appropriate for small areas, a blanket is typically composed of a number of separate electrical/mechanical de-icer segments. Each electrical de-icer segment is long and narrow, and multiple segments are butted against each other to increase coverage of the area to be protected. By creating alternate electrical segments, each fired by separate, isolated electronics, blanket level redundancy may be achieved. Should a portion of a blanket of this type fail, the adjacent segment will shed ice in the failed section at a reduced efficiency. Figure 4A-4 depicts an airfoil with a de-icer blanket configured chordwise with three de-ice zones. Each segment, as well as the combined blanket assembly, is capable of accommodating moderate compound curves, as well as surface twisting and flexing. Blankets are designed to have all electrical connections made at one end of each segment with power leads routed internally within the deicer and each segment connected to a common return. This keeps harness connections to a minimum length and weight and also simplifies harness installation in restricted structures.

The most common installation is the external surface mount where the blanket is bonded to an existing leading edge surface. With this type of installation, the edges of the blanket are tapered to provide a smooth transition from the de-icer blanket to the airfoil surface. The blanket can also be fabricated with a square edge which allows it to be bonded into an existing airfoil recess and results in a non-intrusive installation. Another installation option is the integrated composite leading edge assembly. In this case the blanket is manufactured with a composite structural backing which is designed to meet the structural requirements of the particular application. This option results in a "stand-alone" composite leading edge assembly with the de-icing function built-in and is the most desirable for aerodynamic smoothness.

4A.2.2 Energy Distribution Module

This unit distributes a high voltage, high current, narrow-width pulse to blanket de-icer segments via gating circuits and multiple cables. The gating circuits may be electro-mechanical switches or silicon controlled rectifiers (SCR). A typical electro-mechanical distribution switch has the ability to operate 12 de-ice segments. A similar solid state SCR distributor bank with 12 outputs would be much larger and heavier. A lighter and more compact solid state distributor is under development.

The Energy Distribution Module is a high voltage/high current carrying device; thus wire sizing and run distance is quite critical between each Energy Storage Module and its family of Energy Distribution Modules and blankets. The ohmic drop and inductive reactance in the wiring must be small compared to the impedance of each blanket segment. The module is usually located within a few feet of the blanket lead exits to minimize weight.

Multiple cables connect associated blanket segments to their Energy Distribution Module which is connected by high current disconnects to a single run of low inductance cable leading to its associated Energy Storage Module. Multiple high current disconnects along this single run of low inductance cable can be used to daisy chain Energy Distribution Modules.

4A.2.3 Energy Storage Module

This unit is an electronic driver assembly that stores the blanket firing energy in capacitors and, as directed by the controller sequencing logic, fires the high voltage pulse that will be directed through switching circuits to various blanket de-icer segments via the Energy Distribution Modules. Voltage levels vary between types and required coverage but can be on the order of 200 VDC for simple minimum systems up to near 2000 VDC for more extensive complex systems.

4A.2.4 Controller Module

The controller receives pilot and optional ice sensor inputs, and contains the logic circuits for monitoring and self-test functions. On some types, aircraft power is inputed to the controller and converted to capacitor charging current. On other types, this function may be part of the Energy Storage Module. The wiring run distance between the Controller Module and each Energy Storage Module is not critical and a single run of shielded cable can contain both heavy-current/low-inductance power lines, and control signal lines. Depending upon redundancy requirements, one Controller Module can monitor and operate all EEDS system hardware. For very simple systems, the Controller, Energy Storage, and Energy Distribution modules can be combined into a single assembly.

III.4A.3 USAGES AND SPECIAL REQUIREMENTS

4A.3.1 Airfoil and Leading Edges

The Electro-Expulsive De-Icing System can be adapted to virtually any airfoil or leading edge. Blankets can be fabricated with single or multiple de-icing segments in various widths and lengths and for either chordwise or spanwise installations. Installation is done in much the same manner as that of the standard pneumatic de-icer and can be accomplished either as original equipment or as retrofit. In the latter case, consideration must be given to the effect on structural integrity of any cabling penetrations. Wiring, harnesses, and module placement differs between manufacturers.

Tests have demonstrated continuous (cyclic) ice shedding in all icing conditions when the accreted ice thickness is between 0.08 and 0.10 inches. As with most mechanical ice removal systems, the thinner accreted ice does not shed completely. Residual ice left after the first de-ice cycle is usually removed on the second de-icing sequence. The EEDS has also demonstrated its ability to maintain shed ice particle sizes less than 0.25 inch equivalent spherical diameter (reference 4A-2).

4A.3.2 Windshields

EEDS blankets are not optically clear and so are not appropriate to protect windshields from icing.

4A.3.3 Engine Inlet Lips and Components

Elastomeric surfaced EEDS blankets can be designed and constructed to conform to most engine inlets. Consideration must be given to local radii of curvature when internal conductor patterns are defined. The metallic surfaced blanket with compound curvature has not yet been developed.

The major operational concerns to be addressed are particle size, redundancy level and potential foreign object damage to the engine from a damaged blanket. In a traditional bleed-air thermal system, the entire inlet duct subject to ice accretion or runback is heated to prevent the formation of ice. The water created then must pass through some portion of the engine. In an EEDS protected engine, particles of ice are generated that must also pass through some portion of the engine.

In a traditional bleed-air thermal system, redundancy is achieved by cross-ducting hot air from one engine to the other after a single failure. For EEDS, reasonable redundancy can also be achieved by using interpolated segments. Since ice removal is somewhat less efficient after the first failure of an interpolated blanket, allowance must be made for the thicker ice shed and larger particles produced by this shedding. The reduced shedding efficiency generally means minimum shed thickness on the order of 0.02 inches in the failed area of blanket.

4A.3.4 Turbofan Components

EEDS blankets require electrical connections and have minimum radii of curvature and so are not an appropriate method of protecting moving turbofan components. They can be applied to stationary surfaces as long as the ambient temperatures are compatible with the elastomers used in the blankets. Special elastomers can be used in high temperature locations but they require non-standard manufacturing and test procedures.

4A.3.5 Propellers, Spinners, and Nose Cones

Electro-expulsive ice protection can be applied to propellers but the blankets must be specially designed to withstand erosion, centrifugal loads, and the blade flexing. Centrifugal force actually assists in ice shedding. It typically reduces blanket coverage to those blade radii where the "G" force is less than 3000.

Configuration of an EEDS propeller de-icer can be similar to that of an electrothermal propeller de-icer in thickness, area, and installation. The energy storage module and controller module are located on the non-rotating side of the hub and the energy distribution module on the rotating side. Interface between the non-rotating and rotating sides is a slip ring (see below) assembly which is mounted to the spinner hub. The slip ring is similar to those used for electrothermal systems except that it is rated for higher voltages. It is important to locate the energy storage module as close as possible to the distribution module(s) and blanket(s) to minimize line losses.

Spinners require custom blanket designs to conform to their usually unique shapes. The same electronics used to operate a propeller expulsive system can operate a spinner blanket. The operating environment is usually less severe than that of the propeller.

Nose cones that are required to withstand high temperatures are usually not appropriate for expulsive ice protection. If the nose cone must provide radar or optical transparency then expulsive blankets are not appropriate. Excepting these limitations, expulsive blankets applied to nose cones have characteristics similar to spinner blankets.

4A.3.6 Helicopter Rotors and Hubs

Electro-expulsive de-icing for helicopter rotors and hubs is presently in the concept stage only. Careful consideration must be given to this application because of the increased number of de-ice zones as well as to fatigue due to the higher centrifugal forces. Helicopter blades flex in a complex manner and blankets must be designed to accommodate this motion. When blades are hinged or include blade folding the harness that carries pulse current to the blades must be very carefully designed in order to provide long reliable life in its installed environment. Rotating electronics are required as the high pulse currents cannot be transmitted through slip rings. The Energy Storage Module and the Energy Distribution Module can be integrated into a single package but the additional weight must be added to the rotating system. For rotors and propellers, a synchronizing signal can

be applied so that ice shedding occurs when particle trajectories will not intersect aircraft structures. Additionally, design can provide for simultaneous symmetrical shedding on corresponding portions of the rotating airfoil.

4A.3.7 Flight Sensors

Electro-expulsive protection for aircraft flight sensors is not suitable. While blankets can be applied to a pitot sensor, a static port requires heat to keep its internal section from filling with water or ice. In general, protecting a small and delicate flight sensor can best be accomplished by thermal means.

4A.3.8 Radomes and Antennas

In general, the conductors embedded in expulsive blankets are opaque at radar and radio frequencies. Thus expulsive blankets cannot be used to cover those portions of radomes that must be transparent to radar frequencies or to surround those portions of an antenna that must radiate. In fact, to use expulsive blankets in close proximity to radiated fields, a careful analysis must be performed to ensure that side lobes and fringing effects do not harm intended operation.

4A.3.9 Miscellaneous Intakes and Vents

Electro-expulsive blankets can be used on intakes and vents with the same guidelines and restrictions previously discussed for engine inlets. Since failure modes are less significant than for an engine inlet, some of the restrictions can usually be relaxed to produce a less expensive solution.

4A.3.10 Other

Electro-expulsive blankets can be installed in areas that must be routinely accessed but are subject to freezing rain or other forms of ground icing. Latches, access doors, and inspection ports are typical examples. The blankets are fired manually when access is required during or after icing conditions.

III.4A.4 WEIGHT AND POWER REQUIREMENTS

Weight and power requirements will vary depending upon the manufacturer and the application design. The effects of weight and balance should be considered as part of the application selection.

Careful harness design for expulsive systems is a must to avoid significant weight penalties. Larger systems (150 square feet of protected surface) generally incur a smaller proportional weight penalty than medium sized systems (30 to 150 square feet). Harness weight is usually not an issue for small systems (less than 30 square feet).

III.4A.5 ACTUATION, REGULATION, AND CONTROL

Several methods of actuation and control are possible depending on the level of sophistication desired.

In its simplest form, the pilot would activate the system through a cockpit control switch. The de-icing system would in turn sequence through its de-icing cycles at a preset rate. With this method it must be realized that not all aircraft surfaces are visible to the pilot.

In a more sophisticated state, the Controller Module would receive input signals from an icing rate detector and automatically select a firing cycle based on the icing condition. With this method it is necessary for the designer to be certain that ice accretion at the sensor installation site is representative of the most critical areas to be protected. The same signal can be used to notify the pilot when icing conditions begin. A trade-off must be made to determine the operational requirements with respect to weight, cost, and pilot involvement.

For systems that can tolerate modest accretions of ice, some manufacturers can offer an integral distributed ice detector which uses the blankets and their characteristics to monitor the presence of a significant ice accretion. By monitoring the structural response of blankets with integral sensors, a manufacturer can analyze the natural modes and damping terms of the blanket response

when it is fired. When sufficient ice accretes, these change such that an average ice thickness determination can be made. These sensors are also self-deicing. This type of ice detection is largely independent of the airfoil sub structure characteristics.

A self-test mode can be included in the Controller Module which can either be pilot initiated or automatically initiated. The test would cycle through all the de-icer segments, check the distributor positioning, de-icer circuit integrity and verify the capabilities of the Energy Storage Module. Any deviation from the functional requirements would activate a cockpit warning light.

III.4A.6 OPERATIONAL USE

A system pre-flight checkout is recommended. This checkout can be conducted in either of two ways. The first is through the controller self-test mode which automatically cycles every de-ice zone and monitors circuit integrity as well as that of the Energy Storage and Distribution Modules. This pre-flight checkout method can also be used as an inflight system check. The second method of pre-flight checkout is more readily applicable to smaller electro-expulsive systems. This checkout is very simple where one places their hand on blanket surfaces to ensure that each blanket segment is firing. Also, audible differences are evident for faulty segments.

There is no minimum or maximum ice thickness required or recommended for activation. Operationally, the system should be manually activated in accordance with existing FAA regulations which call for turn-on whenever visible moisture is present and the temperature is below 50 °F (10 °C). Simple systems might merely have a power on/off switch. More complex systems might have a off/auto/manual-on/self-test selector switch plus display of system status and icing rate. In ON and AUTO modes the system will cycle continuously on a pre-determined time basis until the system is placed in the OFF mode. In MANUAL-ON mode the system will operate for one complete cycle of all respective de-ice zones. The pre-determined cycle time is a matter of requirement and designed logic circuits. The firing rate of each segment is controlled by the maximum ice particle size that is desired by the systems designer. A leading edge that accretes ice more rapidly and an engine inlet that must expel only small particles of ice will be fired more frequently than other areas that might accrete ice more slowly and do not present a structural impingement problem. Typically, 1-minute and 3-minute cycle times are used but an EEDS system has the capability to operate with different cycle times assigned to different de-icer segments.

III.4A.7 MAINTENANCE, INSPECTION AND RELIABILITY

There is no scheduled maintenance for an electro-expulsive system. Lack of operational and service experience precludes accurate estimates of maintenance intervals and reliability.

Reliability of the system depends on the complexity of the electronics. In general, the electronic reliability is measured in tens of thousands of power-on hours. Blanket operational life is a function of firing frequency and environmental exposure, e.g., erosion and fluid exposure. Blankets can be expected to provide 250,000 cycles. Thus if every segment is fired once per minute when the system is powered, operational blanket life is 250,000 minutes or 4166 hours in icing conditions.

Periodic visual inspection of blanket surfaces is recommended for detection of weathering, foreign object damage or fatigue cracks. Small nicks or cuts can usually be repaired "on aircraft" thus preventing aerodynamic penalties from surface roughness and preventing small flaws from growing. Should a de-icer segment be damaged or fail, the erosion layer is removed, a replacement segment installed, and a new top layer installed to complete the repair.

No routine maintenance is presently required for the controller, energy storage or distribution modules. All modules should be designed as line replaceable units and accessible for repair or replacement. To operate at extremely low temperatures, non-electrolytic (metallic) capacitors are required to ensure no performance degradation.

III.4A.8 SAFETY

EEDS blankets are designed so that failure will not create a hazard. The energies required to operate a segment are substantial (short duration high current pulses). When a segment fails it can either open or short internally. The open failure prevents capacitor discharge and the Controller Module moves on to the next working segment automatically. When there is an internal short, impedances are so low that the short will always generate a good deal of local heat. The heat vaporizes the internal conductors in the vicinity of the short and causes an open circuit. Materials and blanket geometry have been chosen such that the vapors from the vaporized metal and carbonized polymer are contained within the blanket layers.

The Controller Module is designed to isolate high voltage from aircraft ground. Thus no single failure subjects personnel to hazardous voltages. Most practical EEDS would incorporate ground fault current detection to protect personnel. In order to prevent static charges on blankets from causing internal arcing or creating radio noise, controllers routinely reference the charging circuits to air frame through a large resistance (to maintain personnel safety).

III.4A.9 EMI CONSIDERATIONS

EMI considerations for EEDS blankets, harness, and controllers are significant. Blankets are designed to largely cancel the electric and magnetic fields generated at a relatively small distance from the blankets. Thus MIL-STD-461/462 emissions requirements are satisfied. Harnesses between the Energy Storage Module and blankets generate pulsed magnetic fields that can couple to sensitive wire bundles if those bundles are too close to the high currents. Distances exceeding 6 inches usually afford adequate protection.

Controller Modules are designed to provide compliant operation. They are neither susceptible nor excessively noisy electrically, except for the high current pulse emanating from the Energy Storage Module-to-blanket harness when a segment is fired.

III.4A.10 PENALTIES

Expulsive ice protection sheds particles of ice. If these particles are not tolerable then expulsive ice protection cannot be used.

If radar cross-section is to be controlled then expulsive blankets may not be compatible with radar signature requirements. The high copper content in an expulsive blanket makes it radar reflective therefore potentially compromising to stealth aircraft.

Particle size is a function of firing frequency and the ice accretion rate. To make smaller particles, the blankets must be fired more frequently and this will shorten blanket life.

III.4A.11 ADVANTAGES AND LIMITATIONS

Advantages of the EEDs system are:

a. Low power requirements, so low that it may be operated in all flight regimes including take-off and landing without compromising engine performance.

b. Affords ice protection without creating water run-back and re-freeze.

c. Reduces aerodynamic penalty during icing encounters. Reliable thin ice removal capability with limited residual ice. Ice thickness of 0.08 to 0.10 inches can be consistently shed while maintaining ice shed particle sizes on the order of 0.25 inches sphere-equivalent diameter.

d. Blankets can be triggered to fire at a precise instant thus ice shed from rotors and propellers not impacting aircraft structures is possible.

e. External surface mount installation is easily retrofit to existing aircraft surfaces.

f. Integrated leading edge composite installation is non-intrusive for aerodynamic smoothness.

Limitations of the EEDS system are:

a. System not presently installed or certified on any aircraft.

b. Field service data on maintenance and reliability not available.

c. Some residual ice will remain after cycling.

d. Noise associated with pulsing the system has to be considered.

e. Composite blanket surfaces not as durable as metal surfaces.

f. Lack of blanket transparency to electromagnetic wavelengths.

g. Pulse firing creates some degree of EMI and suppression must be considered.

III.4A.12 CONCERNS

In view of the lack of operational experience with EEDS systems, fatigue of the de-icer surface and conductors is a concern. Laboratory testing has demonstrated over 300,000 impulse cycles without fatigue failure. Fatigue will be a major design goal for rotating applications requiring ice protection.

Electromagnetic interference may occur when the high energy pulse (high voltage) is discharged. The rapid discharge of up to 2000 volts creates transient electromagnetic fields and may cause undesirable signals in communication, control, or navigation equipment. EMI testing of flight test hardware has demonstrated that this undesirable effect can be suppressed.

Whenever a high pulse current harness is installed, care must be exercised to exclude sensitive wiring from the immediate vicinity of the harness. Usually, six inches of spacing is sufficient to limit coupling effects to acceptable levels.

III.4A.13 REFERENCES

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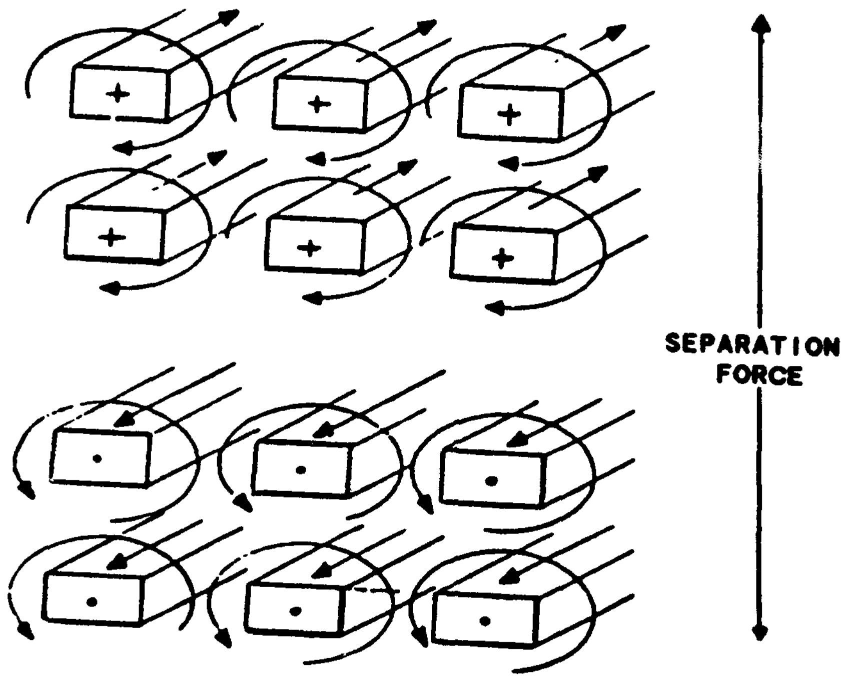

FIGURE 4A-1. EEDS SEPARATION FORCE CONCEPT

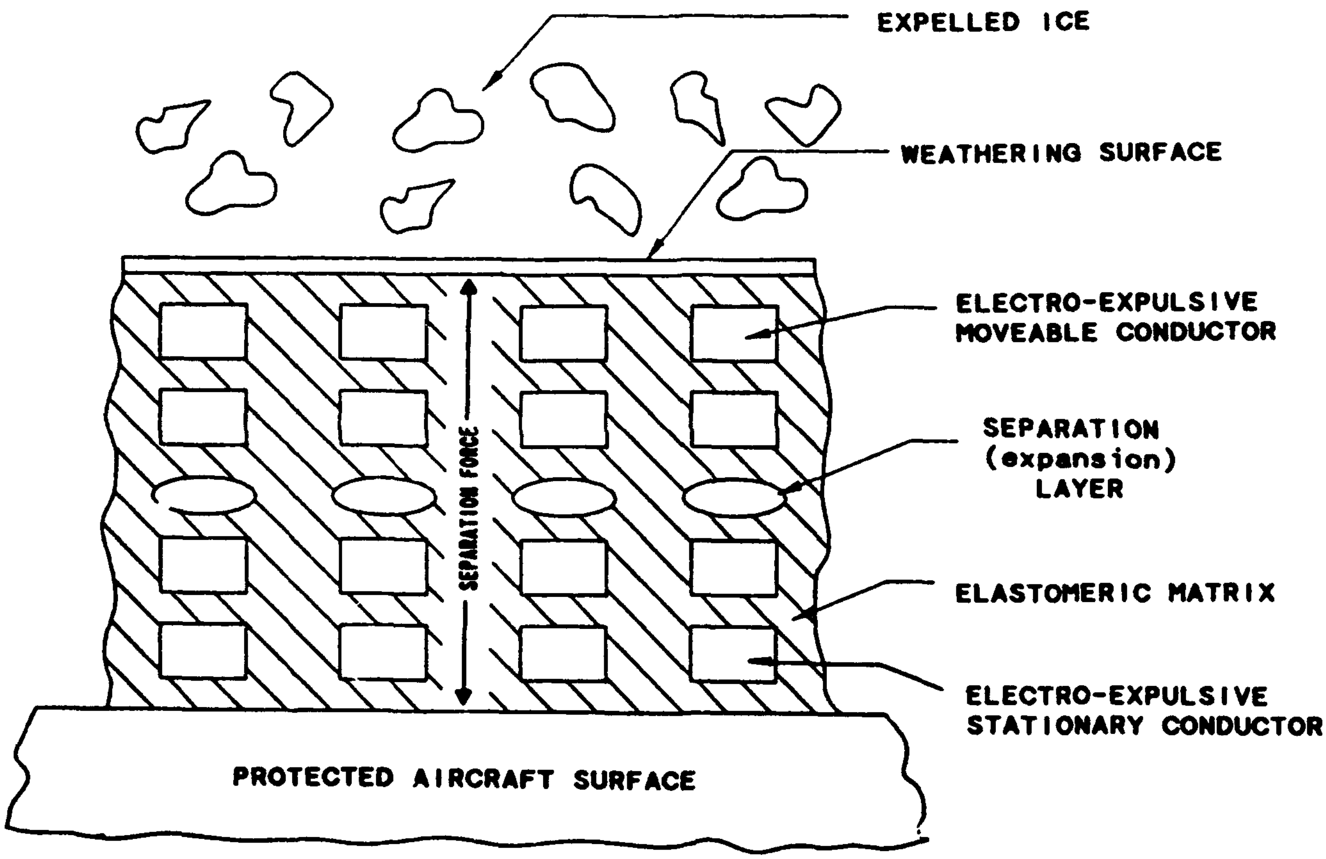

FIGURE 4A-2. CROSS SECTION OF ENERGIZED BLANKET DE-ICER SEGMENT

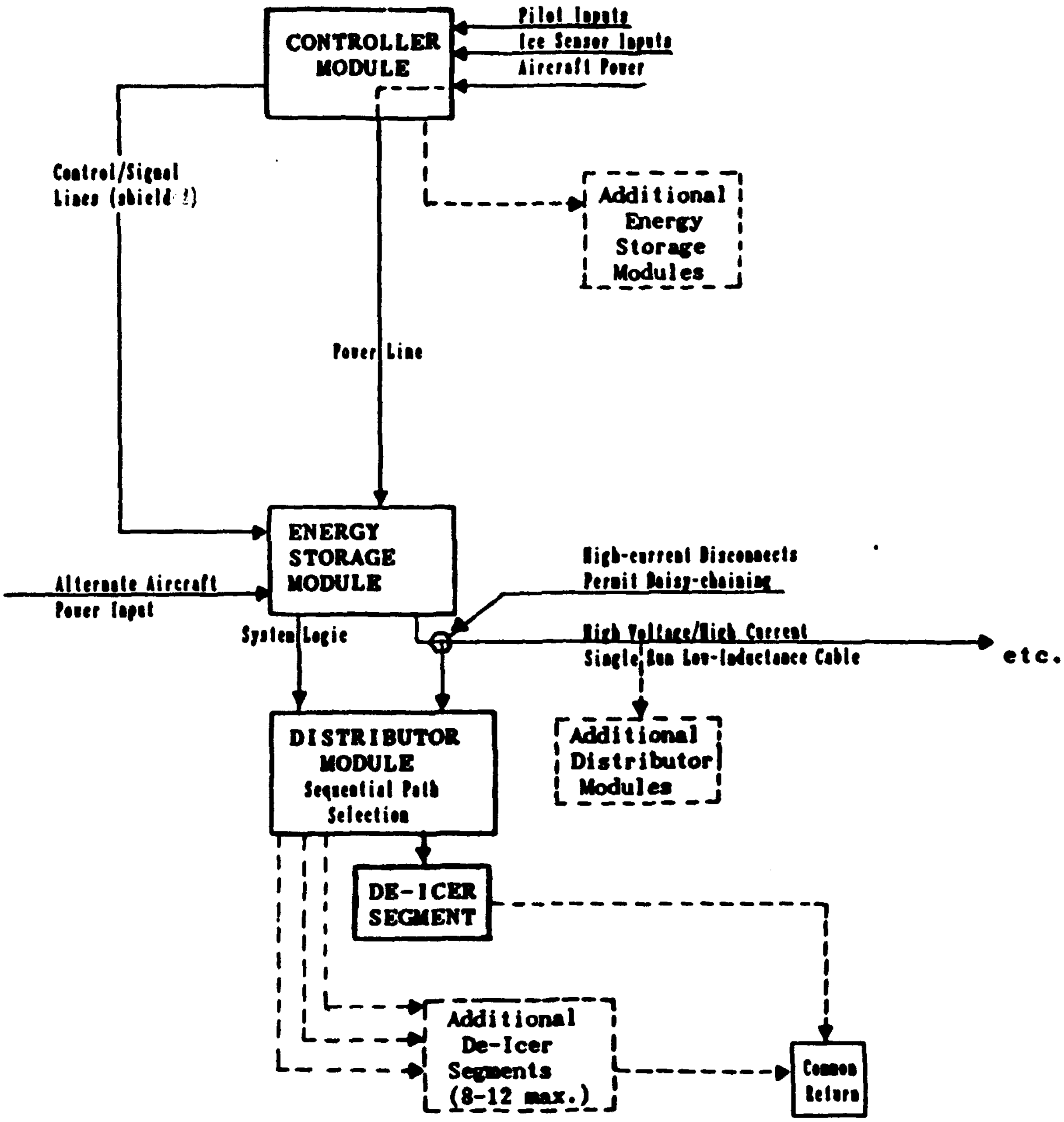

FIGURE 4A-3. TYPICAL EEDS PRIMARY COMPONENTS

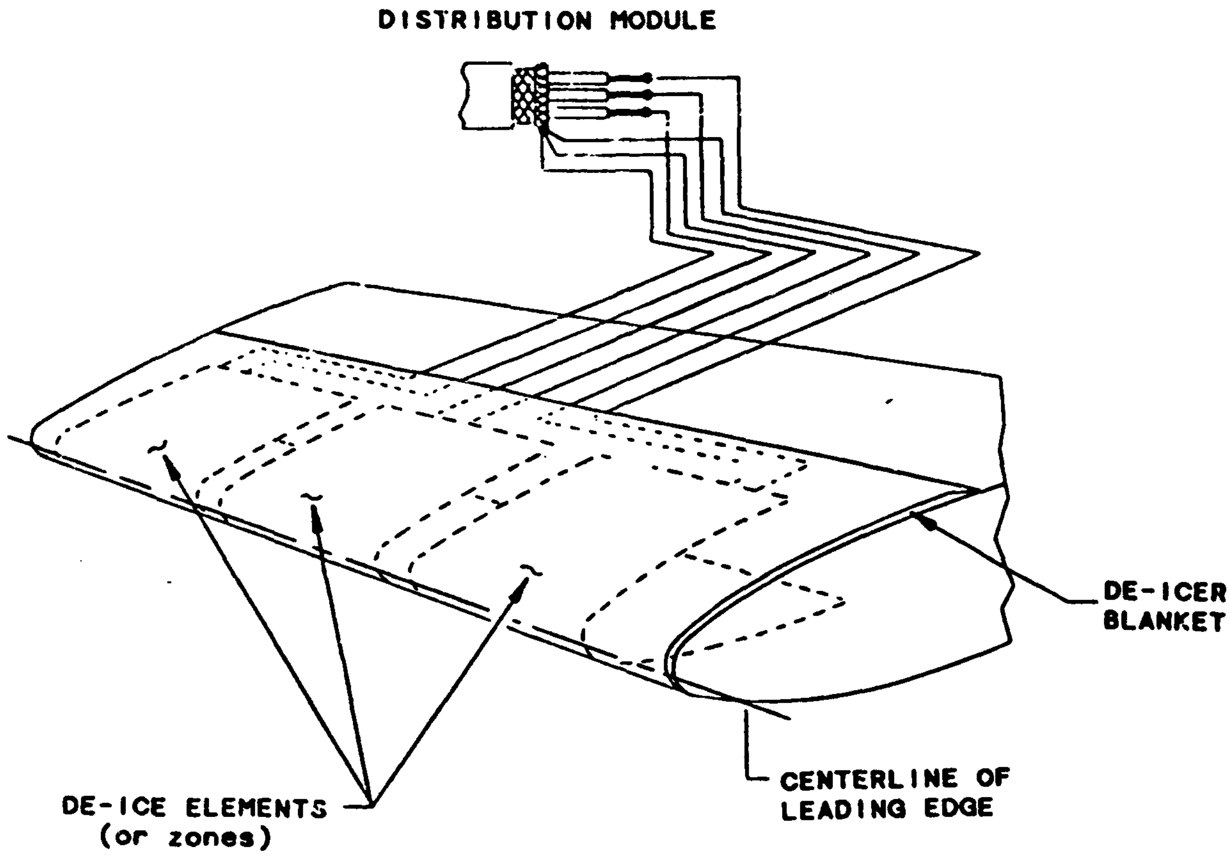

FIGURE 4A-4. EEDS AIRFOIL BLANKET WITH CHORDWISE DE-ICER ZONES

CHAPTER III
SECTION 4B.0
EDDY CURRENT DE-ICING SYSTEMS

CHAPTER III - ICE PROTECTION METHODS

CONTENTS

SECTION 4B.0 EDDY CURRENT DE-ICING SYSTEMS

LIST OF FIGURES

SYMBOLS AND ABBREVIATIONS

Symbol	Description
AC	Alternating Current
°C	Degrees Celsius
DC	Direct Current
ECDS	Eddy Current De-Icing System
EEDS	Electro-Expulsive De-icing System
EIDI	Electro-Impulse De-icing System
EMI	Electromagnetic Interference
°F	Degrees Fahrenheit
FAA	Federal Aviation Administration
SCR	Silicon Controlled Rectifier
V	Volts
VAC	Volts Alternating Current
VDC	Volts Direct Current

GLOSSARY

capacitor - A storage device for electrical energy consisting essentially of two conducting surfaces separated by an insulating material. A capacitor blocks the flow of direct current and effectively permits the flow of alternating current.

eddy current - Current induced in the body of a conducting mass by any variation in the magnetic flux surrounding the mass.

elastomeric - Any substance having the properties of rubber.

electromagnetic interference (EMI) - The field of influence produced around a conductor by the current flowing through it which contributes to a degradation in performance of an electronic receiver. Also called electrical noise, radio interference, and radio-frequency interference.

electro-thermal - Electrical-resistance-generated heat used to evaporate or melt impinging cloud droplets.

impedance - The total opposition (i.e., resistance plus reactance) a circuit offers to alternating current at a given frequency.

inductance - Property of a circuit that tends to oppose any change of current because of the magnetic field associated with the current itself. The unit of inductance is the "henry"

inductive reactance - The opposition to the flow of alternating current as measured in ohms due to the inductance of a circuit.

neoprene - Any of a group of synthetic rubbers. A non-conductor of electricity and superior to rubber in wear resistance.

ohmic drop - A drop (loss) of an electrical current's ability to do work as measured in ohms due to the resistance of a wire or circuit.

phenolic (material) - Any one of several thermosetting plastic materials available which may be compounded with fillers and reinforcing agents to provide a broad range of physical, electrical, chemical, and molding properties.

GLOSSARY (CONTINUED)

planar coil - A number of turns of wire lying essentially in a single plane and within a form made of insulating material. The wire turns introduce inductance into the electric circuit and produce a magnetic flux.

polyurethane - A strong plastic resin that resists fire, weathering and corrosion. A non-conductor of electricity.

runback (ice) - The term given to ice formed from the freezing or re-freezing of water leaving electro-thermal ice protected surfaces.

silicon controlled rectifier (SCR) - A semiconductor device that functions as an electrically controlled switch for DC loads. Also known as a "thyristor."

single-phase (circuit) - An alternating current circuit energized in such a way that the potential between two (or all pairs of) points of entry are either in phase or 180 electrical degrees out of phase.

three-phase (circuit) - A combination of circuits energized by alternating current where the potential between three points of entry differs in phase by one-third of a cycle (120 electrical degrees).

III.4B.0 EDDY CURRENT DE-ICING SYSTEMS

III.4B.1 OPERATING CONCEPTS AND COMPONENTS

Eddy Current De-Icing Systems (ECDS) are classed as electro-mechanical ice protection systems. Accreted ice is expulsed from blanket-protected structures by a strong, rapid outward thrust of the blanket surface. This impulse movement is in reaction to electrical current being pulsed through flattened planar coils embedded within and spanwise along the leading edge of the surface to be protected (figure 4B-1). Over the coil, and separated by an insulation layer, is a conductive target material; over the target layer is a surface erosion layer. The large pulsed currents in the coils induce opposite flowing eddy currents in the conductive target material and these opposing electrical currents (eddy current repulsion) cause the target material and the outer surface to momentarily move away from the coil (figure 4B-2). Reference 4B-1A presents a detailed discussion of this eddy current repulsive force and reference 4B-2 describes the ECDS theory of operation more completely. Actual surface movement is minimal but the acceleration is very rapid. The acceleration debonds and shatters any ice accreted on the outer surface layer. Ice removal is accomplished by aerodynamic forces, inertial forces, or gravity.

The rapid movement of the outer erosion surface and the debonding, shattering, and airflow removal of accreted ice are all actions that are quite similar to those produced by Electro-Impulse and Electro-Expulsive de-icing systems; however, the designs that cause the outer surface to accelerate outward are different. In the Electro-Impulse De-Ice system (EIDI), eddy current repulsion is also used (reference 4B-3) but circular, not planar, ribbon coils are individually attached to the aircraft frame, and eddy currents are induced in the aircraft's skin surface. With Electro-Expulsive De-Ice Systems (EEDS), an electrical current is pulsed in opposite directions through closely-spaced parallel conductors, and an electro-magnetic force is created that forces the conductors apart, imparting an outward force to a moveable outer erosion surface.

In addition to the blankets and coils, other functional components in an Eddy Current De-Icing System are:

- aircraft power converter for capacitor charging current (DC),
- capacitor charging control logic and capacitors for energy storage,
- capacitor energy distribution logic and switching circuits to coils,
- cockpit control panel.

Specific configurations can vary between manufacturers and design requirements. For example, in small aircraft the control logic and the energy charging, storage, switching and distribution functions can all be configured within the power converter module and located in the fuselage (figure 4B-3). In larger aircraft, multiple modules can be used for capacitors and switching circuits to establish branches that would minimize the high voltage wiring runs to the coils (figure 4B-4).

Miscellaneous components include high-current, low-inductance coaxial cabling, electrical interface wiring, connectors, and the appropriate circuit breakers and/or fuses. All of the operating equipment can usually be tailored to operate in either 28 VDC, 115 VAC single-phase or 115/200 VAC three-phase configurations.

III.4B.2 DESIGN GUIDANCE

4B.2.1 De-Icer Blanket

De-icer blankets, also referred to as gloves, differ somewhat in construction among manufacturers, but basically the designs consist of one or more flattened planar coils spaced span-wise to cover the surface to be protected, and a thin metallic "target" sheet positioned over the coils but separated by an insulation layer. The outer surface of the blanket may be metallic or elastomeric (polyurethane or neoprene), depending upon the application. The metallic-surfaced blanket is more difficult to install but has better erosion characteristics and fewer maintenance concerns. The elastomeric surfaced blanket is easier to install and more applicable to retrofit applications.

The target sheet is free to move only slightly outward, and is restrained in the spanwise and chordwise direction. An alternative approach would be to configure the blanket layers such that the coil accelerates away from the target material. In this case the target sheet would be attached to the airfoil and the coil (layer) would be free to deflect outward. The de-icing principle would be the same. The target sheet must be made from a highly conductive material to achieve strong eddy currents. Solid copper sheet provides excellent conductivity; aluminum provides about half that of copper. However, neither is extremely pliable. Solid beryllium copper alloy sheet can provide high electrical conductivity and good strength. Woven mesh is less conductive than the materials already mentioned but is extremely flexible and compliant. A tradeoff must be made with respect to weight and thickness when selecting the material and configuration.

Typically, the de-icing blanket is bonded onto the leading edges of airfoils in much the same manner as pneumatic de-icers or electro-thermal propeller de-icers. This technique lends itself well to solid composite or metallic structures, where access is limited to the outer surface. Retrofit operations are also facilitated because modification to the existing airfoil is generally not required, although small access ports in the aircraft surface are needed for the electrical cables.

Installation is more critical for some airfoils than others. An airfoil's characteristics are defined primarily by the shape of the camber line. By adding leading edge material such as the de-icing blanket, the camber line and therefore the airfoil is being modified to something different than originally designed. In the case of larger commercial transports and new commuter and general aviation aircraft which utilize highly optimized laminar-flow wings, the preferred installation would be to supply the blanket as one original equipment unit. Retrofitting the de-icing blanket to a highly optimized laminar section takes careful analysis, design, and installation. The best design criteria here is to design the add-on section as closely to the original leading edge shape as possible and in so doing,

make sure that the new camber line blends in with the old as smoothly as possible and that the leading edge pressure gradient is consistent with the original shape. Surface discontinuities or changes in slope, which may result in boundary layer disturbances and increased drag, should be avoided.

Attaching the blanket to the aircraft with flexible adhesive works well. Adhesives shown to work are Hexcel 3140 urethane and flexible epoxies such as Hysol 9340 or 3M 2216. Hard fasteners are not recommended for most installations due to the high acceleration forces involved. An exception is locations where the glue bond line may be peeled. Adhesives have high shear strength, but little peel strength. For instance, one or two small rivets may be placed at the corners of the bond line through the outer alloy skin and the aircraft surface. The use of bolts should be avoided. Even the head thickness (0.10" or 2.5 mm) of a countersunk AN525/10-32 R7 screw may be unacceptable for critical airfoils.

A small aircraft may have three blanket sections with four coils each: one blanket on each wing and one on the tail. The power supply would automatically pulse all 12 separate coils in sequence at a designated firing cycle (figure 4B-3). Ice removal is most efficient if each coil is pulsed twice in a row. To achieve the thinnest possible cross-section, the coils are typically made from flattened braid wire, or thin foil which has been chemically etched to the coil configuration in a process similar to the manufacture of printed circuit boards. The impedance of the coil must be closely controlled for a given power supply and pulse power cables. For example, the coil used with a 500 volt power supply must have a much lower impedance than a coil with a 2000 volt power supply. Linear circuit theory yields the optimum coil density. Basically, the coil must have enough impedance (turn density) so that the capacitor energy is sufficiently transferred to the coils, but not so much impedance that the system becomes too slow and stops producing the high acceleration forces that remove ice. Higher turn density coils have a higher impedance. Such a coil uses most of the capacitor energy, but may slow the electrical current down too much. The resulting low acceleration of the blanket outer skin will not remove thinner ice. A coil with a low turn density reacts quickly but does not use most of the capacitor stored energy.

4B.2.2 Energy Distribution Module

This module distributes a high voltage, high current, narrow width pulse to the coils in the blanket segments via gating circuits and multiple cables. The gating circuits can be electro-mechanical stepping switches or silicon controlled rectifiers (SCR).

The energy distribution module is a high voltage/high current-carrying device; thus wire sizing and run distance are quite critical between each energy storage module and its family of energy distribution modules and blankets. The ohmic drop and inductive reactance in the wiring must be small compared to the impedance of each blanket segment. The module may be located within a few feet of the blanket lead exits to minimize weight.

Multiple cables connect associated blanket segments to their energy distribution module, which is connected by high current disconnects to a single run of low inductance cable leading to its associated energy storage module. Multiple high current disconnects along this single run of low inductance cable can be used to daisy chain energy distribution modules.

4B.2.3 Energy Storage Module

This module is an electronic driver assembly that stores the blanket-firing energy in capacitors and, as directed by the controller sequencing logic, fires the high voltage pulses that are directed through switching circuits to various blanket de-icer segments via the energy distribution modules. Voltage levels vary among types and required coverages, and can range from around 200 VDC for simple, minimal systems to nearly 2000 VDC for complex, extensive systems. The time required to charge the capacitor is a function of the final voltage, and a "typical value" is approximately two seconds.

4B.2.4 Controller Module

This module receives pilot and optional ice sensor inputs, contains the logic circuits for monitoring and self-test functions, and, in general, is used to direct the operations of the de-icing system. On some types, aircraft power is input to the controller and converted to capacitor charging current. On other types, this function may be part of the energy storage module. The wiring run distance between the controller module and each energy storage module is not critical and a single run of shielded cable can contain both heavy-current/low-inductance power lines and control signal lines. One controller module can monitor and operate all ECDS system hardware, although additional controller modules may be included to meet redundancy requirements. For very simple systems, the controller, energy storage, and energy distribution functions can all be combined into a single assembly.

III.4B.3 USAGES AND SPECIAL REQUIREMENTS

4B.3.1 Airfoil and Leading Edges

The Eddy Current De-Icing System can be adapted to virtually any airfoil or leading edge. The blankets are attached to the airfoil in much the same manner as standard pneumatic de-icer boots. Blankets must always be blended smoothly to any airfoil surfaces to avoid step disturbances in the boundary layer and excess drag. Additionally, the blanket can also be designed and manufactured as a complete leading edge assembly and be installed as such.

Allowable ice thickness must also be considered. The system can prevent the buildup of ice greater than 0.10" (2.5 mm). The pilot (or ice detector) can always turn the system on safely; a threshold thickness does not have to be obtained.

4B.3.2 Windshields

The Eddy Current blankets are not optically clear and thus are not appropriate for windshield ice protection.

4B.3.3 Engine Inlet Lips and Components

Although Eddy Current de-icers have not been installed on production engine inlets, they can be fitted to engine inlet lips and other components such as splitters or guide vanes. Consideration must be given to the design for complex shapes so that the target and coils conform to the contour of the shapes. In this application, the de-icer blanket would likely be formed to the contour prior to installation. Several coils should be placed radially around the inlet. Generally coils are placed in redundant pairs. Typical coil spacing is 18" between each coil. Coils can be placed directly on the ice formation surface or just behind the leading edge for radii sharper than 0.75" (19 mm). The outer metal strip of the de-icer blanket, typically aluminum or titanium alloy, is formed between dies or stretch-formed over a male dye.

Preliminary testing for shed particle size and thickness has been done as part of Air Force/NASA ice protection tests (reference 4B-4). See figure 4B-5. For the two icing conditions included in this test, firing each coil every two minutes or less kept shed particle thickness below 0.10" (2.5 mm). More frequent firings may assure thinner shed ice particles.

4B.3.4 Turbofan Components

Suitability of the Eddy Current De-Icing System for turbofan components has not been evaluated.

4B.3.5 Propellers, Spinners and Nose Cones

The configuration of the Eddy Current propeller de-icer is similar to that of an electrothermal propeller de-icer in terms of thickness, area, and installation (figure 4B-6). However more development testing is needed to establish blanket criteria for withstanding erosion, centrifugal loads and blade flexing. It is probable that the blade and the de-icing system must be integrally designed. The energy storage unit and control module could be located on the non-rotating side of the hub and the distributor on the rotating side. The connecting means between the two could be a slip ring assembly mounted to the hub. The slip ring would be similar to those used with electrothermal propeller de-icing, except that it would be rated for the higher voltage. It is important to mount the energy storage unit as close as possible to the distributor and de-icer to minimize line losses.

Suitability of the system for spinners and nose cones has not been completely evaluated. Non-rotating nose cones can be wired in the same manner as wing leading edge coils. Rotating cones or spinners require slip rings rated for transmitting the high current pulse from the power supply to the coils. Laboratory testing of such slip rings has been done (reference 4B-5). Alternatively, a small capacitor/power supply can be installed within the nose cone as indicated in reference 4B-3.

4B.3.6 Helicopter Rotors and Hubs

Eddy Current de-icing for helicopter rotors and hubs is presently in the concept stage only and more development work is needed. See the discussion of propeller de-icing, Section 4B.3.5.

4B.3.7 Flight Sensors

Eddy Current protection for aircraft flight sensors is not suitable. In general, protecting a small and delicate flight sensor can best be accomplished by thermal means.

4B.3.8 Radomes and Antennas

In general, the conductors embedded in eddy current blankets cannot be used to cover those portions of radomes that must be transparent to radar frequencies or to surround those portions of an antenna that must radiate. In fact, before using eddy current blankets in close proximity to radiating fields, a careful analysis must be performed to ensure that side lobes and fringing effects do not degrade intended operation (references 4B-6, 4B-7).

4B.3.9 Miscellaneous Intakes and Vents

Eddy Current blankets can be used on intakes and vents in accordance with the same guidelines as previously discussed for engine inlets. Since failure modes are less significant than for an engine inlet, some of the restrictions can usually be relaxed to produce a less expensive solution.

4B.3.10 Other

Struts, pylons, wheel covers, tail assemblies, and other aircraft surfaces where ice forms are candidates for the Eddy Current De-Icing System. Blankets can also be installed in areas that must be routinely accessed but are subject to freezing rain or other forms of ground icing. Latches, access doors, and inspection ports are typical examples. The blankets are fired manually when access is required during or after icing conditions.

III.4B.4 WEIGHT AND POWER REQUIREMENTS

Low power requirements for the ECDS have been documented in testing at the NASA-Lewis Icing Research Tunnel (reference 4B-4). Weight and power requirements vary depending upon the application and the manufacture's design. Weight and power requirements both increase with the extent of protection needed (aircraft size, wings, empennage, engine inlets, etc.), and power requirements increase as the firing cycle is shortened. As a very general guideline, weight estimates vary from 50 pounds for minimum applications, to 500 pounds for maximum applications. Power estimates range from 0.015 watts/sq. in. of coverage for a large aircraft using a three-minute firing cycle to 0.7 watts/sq. in. using a one-minute firing cycle.

III.4B.5 ACTUATION, REGULATION, AND CONTROL

Several methods of actuation and control are possible, depending on the level of sophistication desired. In the simplest form, the pilot activates the system through a cockpit control switch. Upon power-up, the system automatically checks for short circuits, ground faults, and open electrical circuits, and then sequences through the de-icing cycles at a preset rate. Although all protected surfaces may not be visible to the pilot, he need not be concerned lest he turn on the system with "too small" an ice buildup, since a minimum ice buildup does not have to occur before de-icing.

In a more sophisticated configuration, the control logic receives input signals from an icing rate detector and selects a firing cycle accordingly. With this method it is necessary that ice accretion at the sensor be representative of the most critical areas to be protected. The same signal can be used to notify the pilot when icing conditions begin. Choice of configuration requires consideration of operational requirements in light of weight, cost, and pilot involvement.

A self-test mode can be included in the control logic which can either be pilot-initiated or automatically initiated. The test cycles through all the system circuitry and any deviation from the functional requirements activates a cockpit warning light.

III.4B.6 OPERATIONAL USE

A system pre-flight checkout is recommended. This checkout can be conducted in either of two ways. The first employs a self-test mode which automatically cycles every de-ice zone and monitors circuit and system integrity. (This method can also be used as an in-flight system check.) The second method of pre-flight checkout is best suited to smaller systems. One places his hand on blanket surfaces to ensure that each blanket segment is firing and also listens for audible differences that should be evident for faulty segments.

There is no minimum or maximum ice thickness required or recommended for activation. Operationally, the system should be activated in accordance with existing FAA regulations which call for turn-on whenever visible moisture is present and the temperature is below 50 °F (10 °C). Simple systems might merely have a power on/off switch. More complex systems might have a off/auto/manual-on/self-test selector switch plus a display of system status and icing rate. In the ON and AUTO modes the system would cycle continuously on a pre-determined basis until the system was placed in the OFF mode. In MANUAL-ON mode the system would operate for one complete cycle of all respective de-ice zones. The pre-determined cycle time is a matter of requirement and designed logic circuits. The firing rate of each segment is controlled by the maximum ice particle size that is desired by the systems designer. A leading edge that accretes ice rapidly, or an engine inlet that must expel only small particles of ice, would require more frequent firing than other areas that might accrete ice more slowly or do not present a structural impingement problem. Typically, one-, two-, and three-minute cycle times are used, but the system has the capability to operate with different cycle times assigned to different de-icer segments.

III.4B.7 MAINTENANCE, INSPECTION, AND RELIABILITY

The lack of operational and service experience precludes a general statement of maintenance requirements or reliability. Periodic visual inspection of blanket surfaces is recommended for detection of weathering, foreign object damage or fatigue cracks. Small nicks or cuts can usually be repaired "on aircraft," thus preventing aerodynamic penalties from surface roughness and also preventing small flaws from growing. If a de-icer segment fails or is damaged, the erosion layer is removed, a replacement segment is installed, and a new top layer is installed to complete the repair.

No routine maintenance of the electronic modules is required. All modules should be designed as line-replaceable units and should be accessible for repair or replacement. Non-electrolytic (metallic) capacitors are required to ensure no performance degradation at extremely low temperatures. Additionally, a temperature switch and heating element can be included in the design so that the capacitor bank energy storage remains constant at temperatures below -40°F (-40°C). For rotary applications, any slip rings used should be periodically inspected for wear.

III.4B.8 EMI CONSIDERATIONS

Laboratory EMI measurements of the ECDS have been made (references 4B-6, 4B-7) and were within the Category A and Z limits of RTCA/DO-160B Section 21.

III.4B.9 PENALTIES

See limitations listed below.

III.4B.10 ADVANTAGES AND LIMITATIONS

Advantages of the EDCS System are:

a. Low power requirement. Power requirements are 30 to 50 times less than for hot air or electrothermal anti-icing systems. Requirements are so low that the system may be operated in all flight regimes, including take-off and landing, without compromising engine performance.

b. Reliable de-icing. Ice of all types - thick, thin, clear, glaze, wet - is effectively removed.

c. Minimum ice buildup. By cycling the system rapidly, on the order of once per minute, ice thickness may be limited to less than 0.10 inch in most conditions.

d. Minimal residual ice. Regardless of the firing cycle, residual ice thickness is less than 0.020 inch (0.75mm).

e. No runback and/or refreezing. This eliminates a concern for some thermal systems regarding ice forming beyond protected surfaces. Advantageous in engine inlet applications.

f. Minimal aerodynamic penalty, during both icing and non-icing conditions. The firing cycle involves only instantaneous intrusion into the air stream during icing conditions. The integrated leading edge composite installation is non-intrusive during non-icing conditions.

g. Pilot judgement not required. The pilot does not have to determine if the ice buildup has reached a "threshold thickness," since the system can be turned on at any time and function effectively.

h. Retrofit application. The system can be retrofit to composite or aluminum aircraft structures. It does not need to be designed inside of the leading edge.

i. Low electromagnetic interference. The system passes RTCA/DO-160B (references 4B-6 and 4B-7).

Limitations of the ECDS system are:

a. New and not certified. The system is not presently certified on any aircraft. See reference 4B-8 for a discussion on FAA concerns for certification.

b. Residual ice formation. As with all mechanical de-icing systems, some ice (0.005" to 0.070") may remain on the leading edge in some conditions. This may be unacceptable for super-critical airfoils.

c. Noise. Noise associated with firing the coils may be discernible in the cabin or cockpit of smaller aircraft.

III.4B.11 CONCERNS

Technical obstacles must be cleared before the ECDS is flown. Some concerns can be addressed with ground testing, minimizing expensive flight tests.

a. Stresses and fatigue induced in the aircraft skin. The coil will induce load and stresses in the airfoil. Special strain gauge testing must be done. Such testing is discussed in reference 4B-5.

b. Stresses and fatigue life of the outer metal strip. Aluminum or titanium alloy outer metal strips experience a small sudden deflection when the system is pulsed. Special strain gauge testing must be conducted.

c. Power supply reliability. Endurance testing of the power supply needs to be done. Vibration testing will be required prior to actual flight.

d. Overall system reliability. To satisfy the aircraft industry and the FAA, endurance testing of the entire system must be performed.

e. Electromagnetic interference (EMI). EMI test results of the ECDS for the FAA are given in reference 4B-6. Good shielding and grounding of coils and cables are the key to obtaining good EMI results, thus special precautions must be taken.

f. Lightning strikes. No lightning strike testing of the ECDS has been done.

III.4B.12 REFERENCES

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4B-2 Smith, S.O., and Zieve, P.B., "Thin Film Eddy Current Impulse Deicer", AIAA 28th Aerospace Sciences Meeting, Reno, NV, January 8-11, 1990, Paper No. AIAA-90-0761.

4B-3 Zumwalt, G.W., Schrag, R.L., Bernhart, W.D., and Friedberg, R.A., Electro-Impulse De-Icing Testing Analysis and Design", NASA Contractor Report 4175, grant NAG3-284, September 1988, general release date September 1989.

4B-4 Bond, T. H.; Shin, J.; Mesander, G. A.; Yeoman, K. E., "Results of USAF/NASA Low Power Ice Protection Systems Test in the NASA Lewis Icing Research Tunnel," NASA TP 3319, 1993.

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4B-6 Zieve, P.B., Ng, J., Friedberg, R.A. and Robinson, C., "Suppression of Radiating Harmonics in Electro-Impulse De-icing System", DOT/FAA/CT-TN90/33.

4B-7 Zieve, Peter, "Electromagnetic Emissions from a Modular Low Voltage EIDI System", AIAA 27th Aerospace Sciences Meeting, Reno, NV, January 9-12, 1989, Paper AIAA-89-0758.

4B-8 Masters, C.O., "Electro-Impulse De-Icing Systems: Issues and Concerns for Certification", AIAA 27th Aerospace Meeting, Reno, NV., January 9-12, Paper No. AIAA-89-0761.

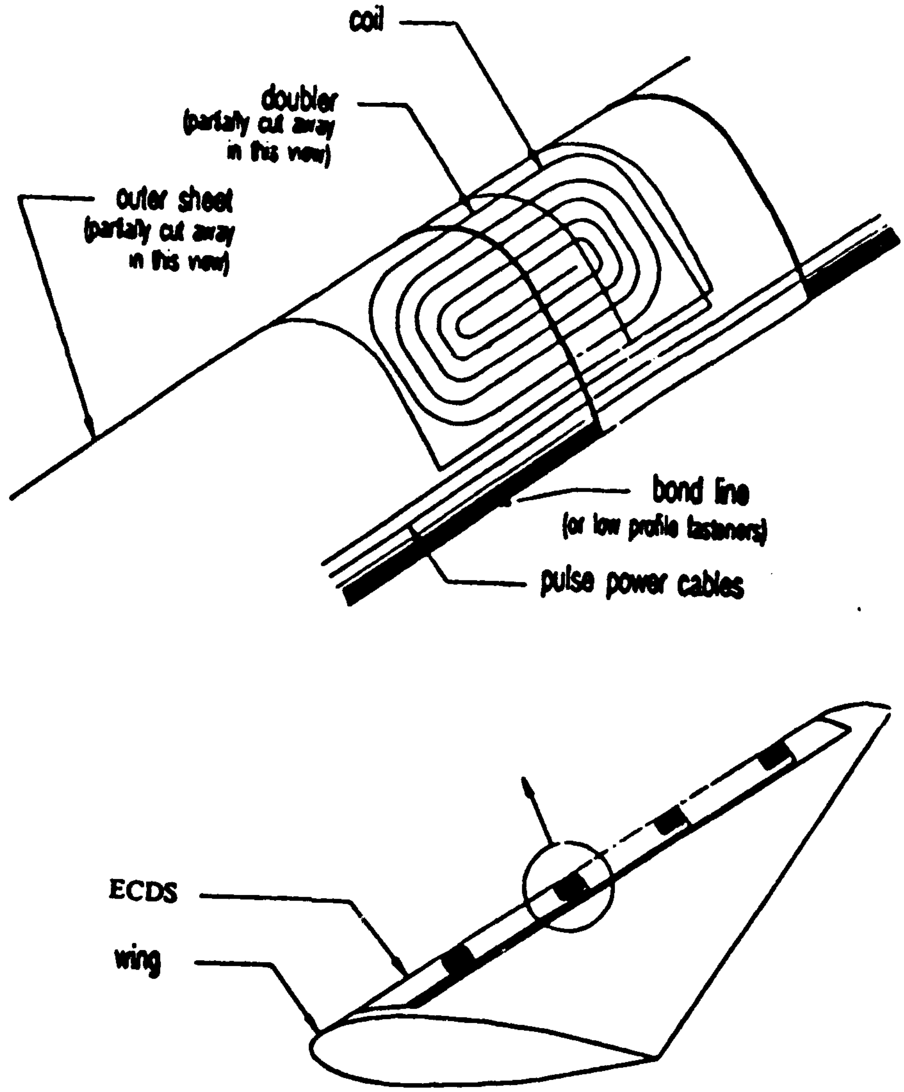


FIGURE 4B-1. ECDS Planar Coil

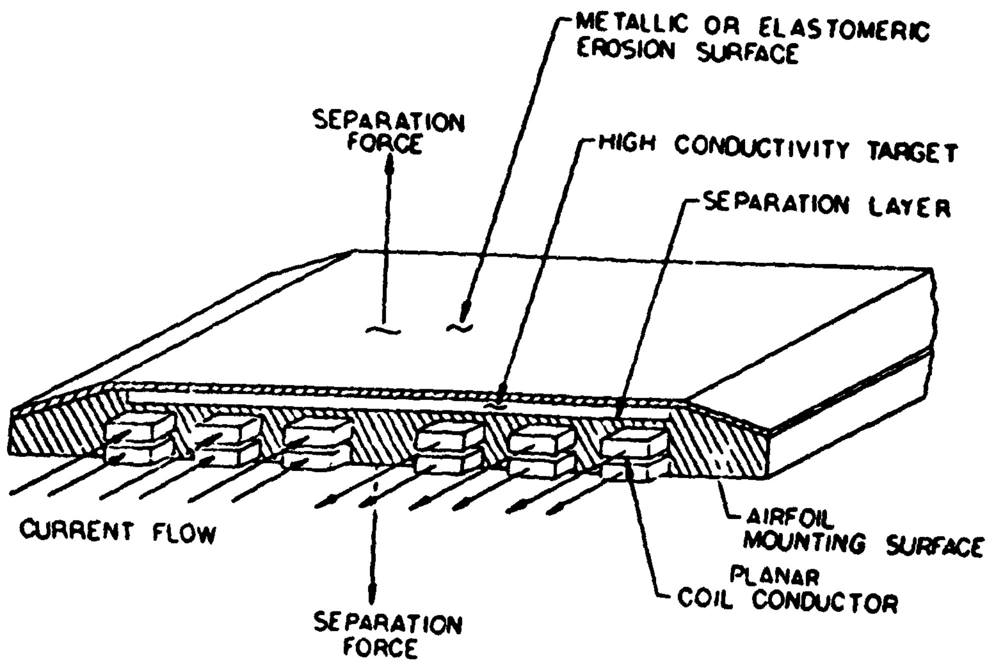


FIGURE 4B-2. ECDS Blanket Assembly

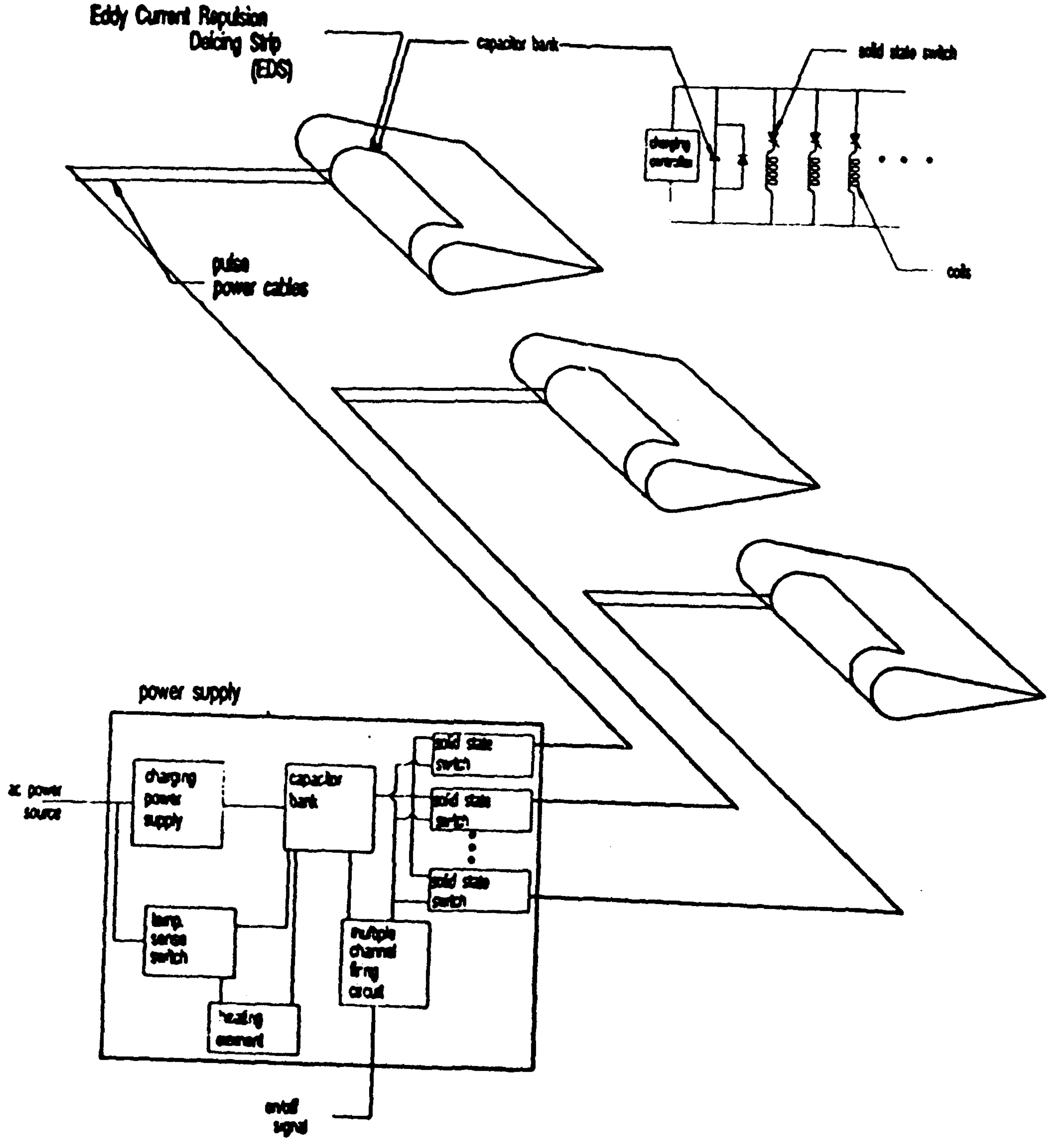


FIGURE 4B-3. ECDS Minimum System

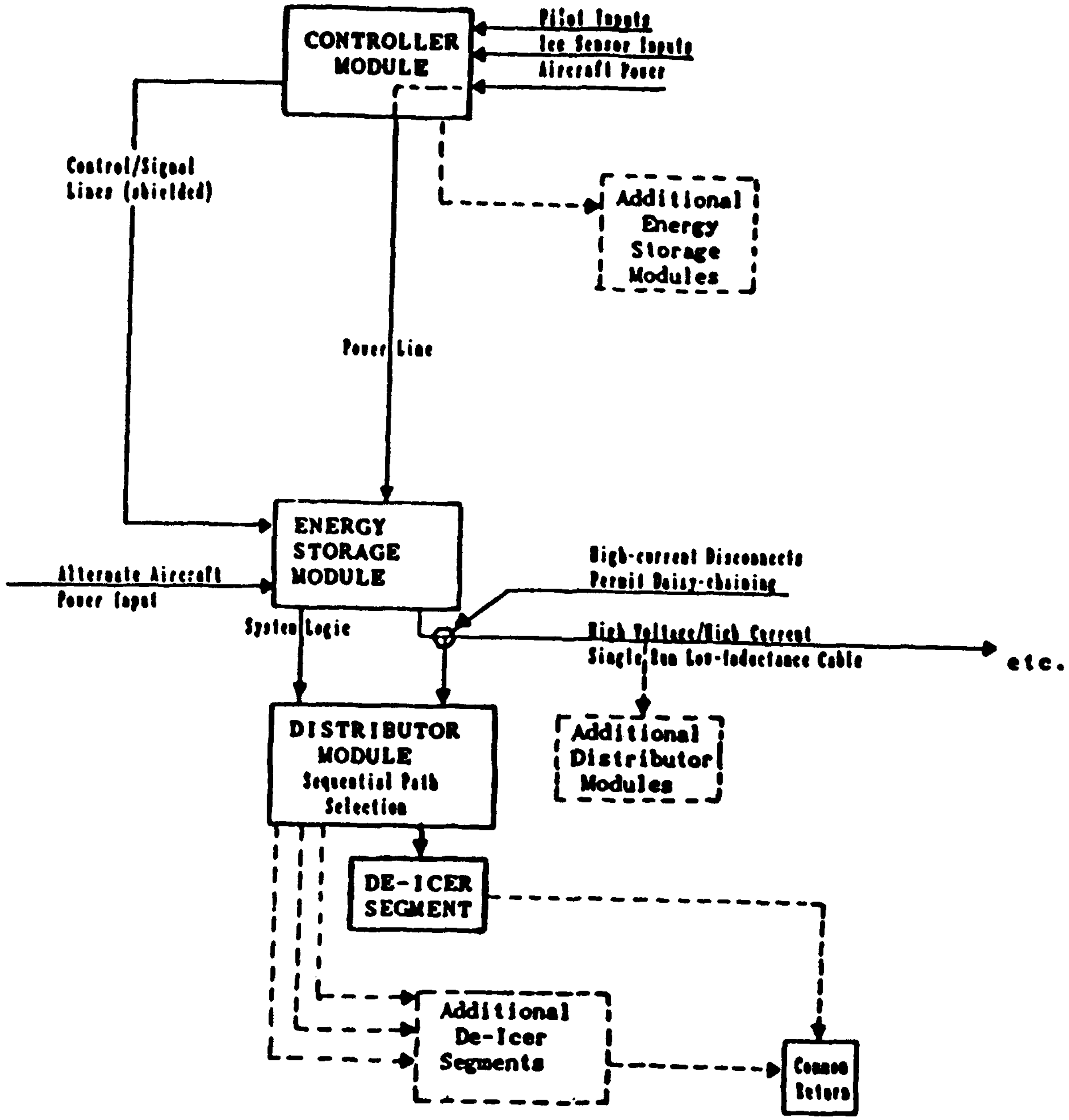
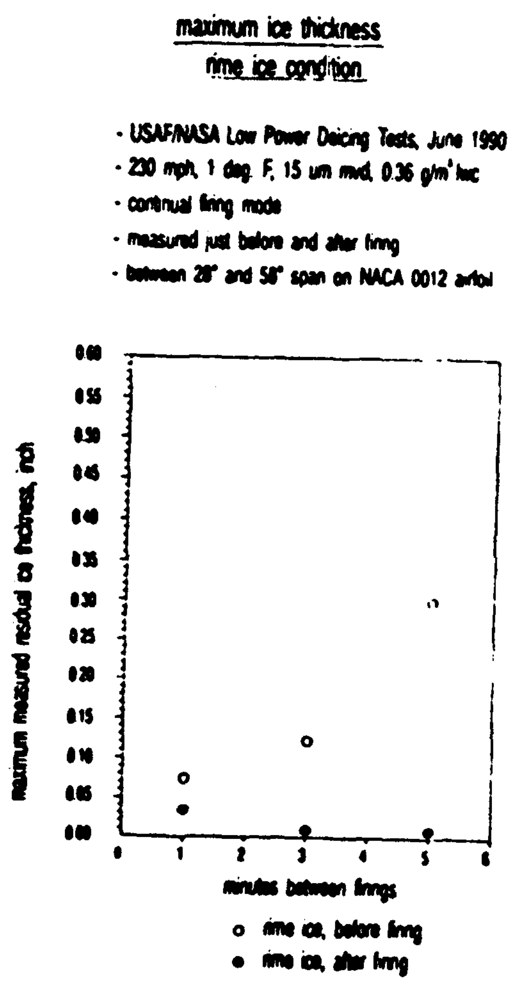


FIGURE 4B-4. ECDS Complex System



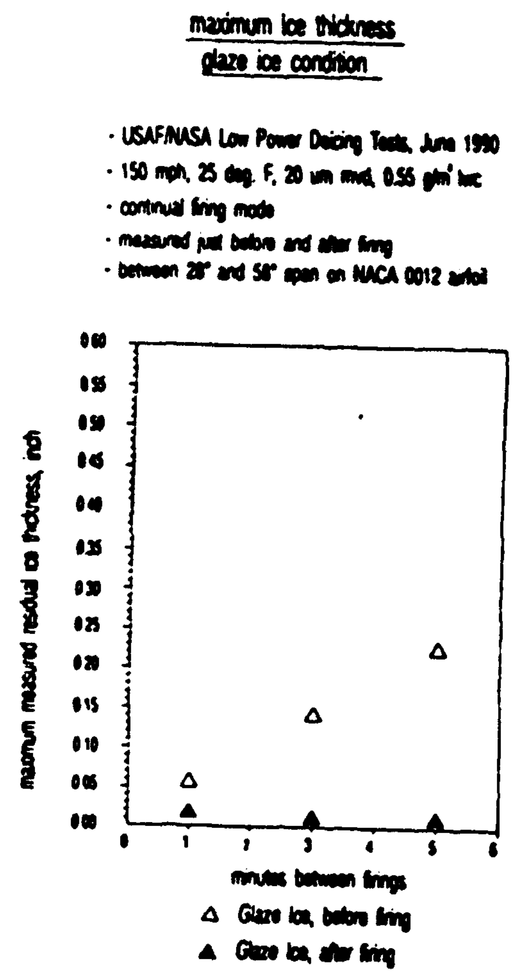


FIGURE 4B-5. Design of ECDS Cycling Time

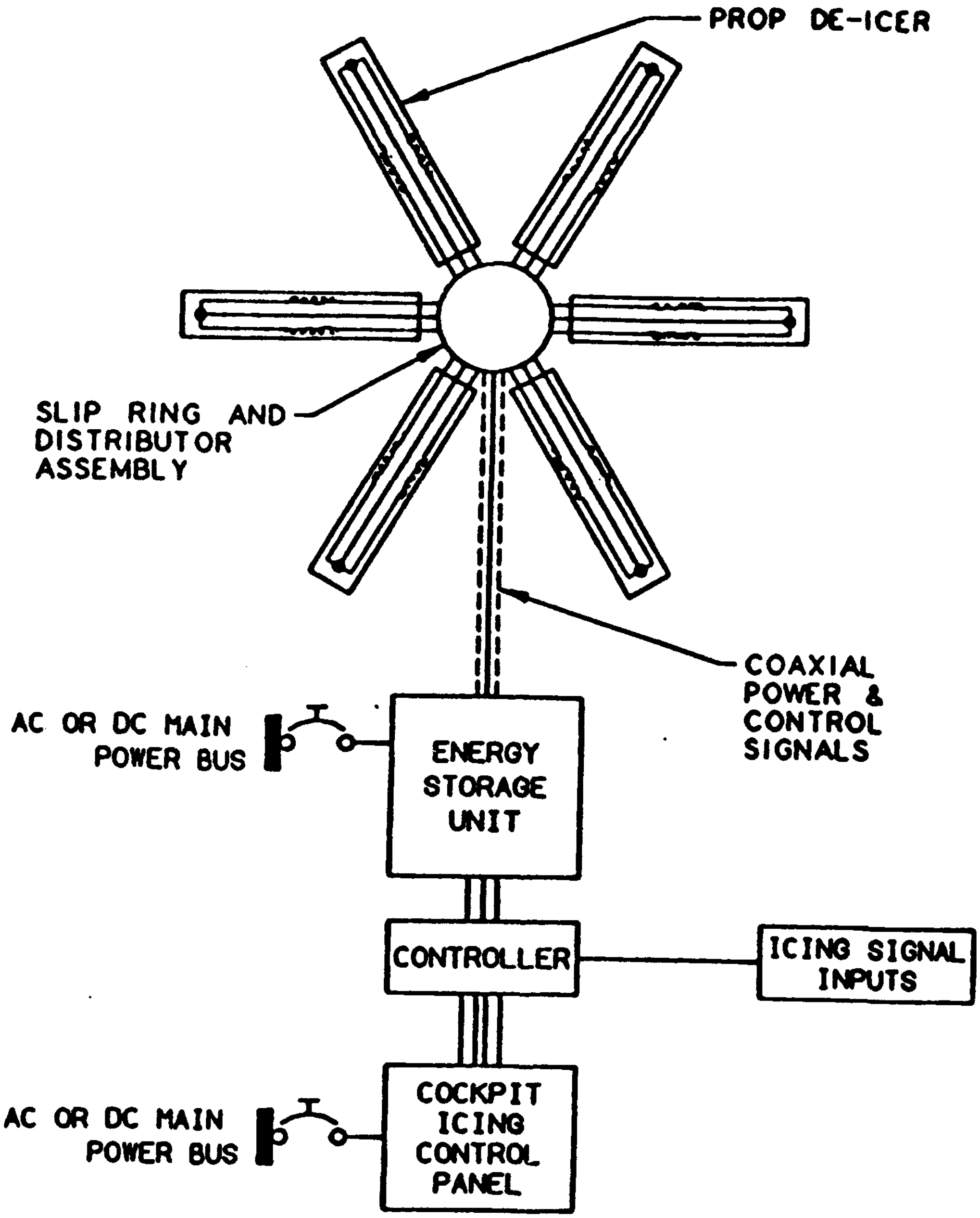


FIGURE 4B-6. ECDS Propeller Block Diagram

5.3.3 Engine Inlet Lips and Components

Ice protection is necessary for turbojet engine inlet lips or aerodynamic surfaces in front of them to prevent engine damage due to ingestion of ice. Decrease of the inlet area due to ice buildup is not a significant factor, except for extremely small engines. Because compressor bleed air is readily available from the engine and its delivery maintenance requirements are low, this air is usually preferred as the heat source. The hot air is usually drawn from the high pressure compressor through a bleed manifold. The anti-ice system ducts this air forward from the engine to the inlet and convectively heats the lip by directing the hot air against the inner surface of the inlet lip.

There exist three basic geometric arrangements for such a convective heating system. The first is called the piccolo tube system (figure 5-34). It uses a hoop of small diameter tubing to distribute the hot bleed air circumferentially. Along the tube periphery, small exit holes are drilled in order to direct the air as jets against the inlet lip skin where it transfers heat to the super cooled water droplets impinging on the outside of the lip skin. The spent anti-icing air then flows aft into the rear chamber through holes in the front bulkhead near the inlet throat. In the rear chamber, warming of the inlet inner barrel occurs before the air is vented overboard through an exit in the outer barrel. Although it is less thermally efficient, this arrangement is particularly useful for thin leading edges (as on supersonic wings) or flush scoops. Cost of manufacturing the single skin system may be lower for many applications.

The second arrangement is called the double-walled system because of the heating channel formed between an inner skin and the exterior skin (figure 5-34). This method introduces the air into the inlet lip "D-duct" (figure 5-34) and it is then forced to flow into the channel between the two walls. Here heat transfer occurs between the hot air and the external skin. In the double-walled system, high circumferential air velocities in the D-duct is necessary to obtain a good distribution around the D-duct. This suggests that an effective heating system can be obtained without the need of a double wall. This new concept is called the "swirl system" (RohrSwirl™).

The third arrangement is the Swirl Anti-Ice System (patent no. 4,688,745). Its swirl nozzle introduces the air in the tangential direction at the center of the D-duct cross-sectional area. The nozzle pumps the air in the lip around the circumferential chamber, producing a swirl flow several times larger than the nozzle flow. The swirl nozzles also act as a flow restrictor as it operates choked at all flight conditions, and its exit diameter controls the flow of hot air needed for anti-icing purposes. The hot air transfers heat to the lip skin, then flows aft into the rear chamber through holes similar to those in the piccolo system. The hot air is then vented overboard through the outer barrel.

A hot gas anti-icing system for the engine inlet lip of a typical FAR Part 25 transport is illustrated in figure 5-34. The area of the nacelle requiring anti-icing extends aft from the leading edge along the inner and outer surface of the inlet lip; a horizontal distance of 6 inches (15 cm) is typical. However, the exact distance may depend on cowl configuration and impingement area. The anti-icing engine compressor bleed air is distributed around the lip by a modified "D" duct and passes through holes in the leading edge of the inner skin into 0.40 inch (10 mm) gas passages. The air flows

aft through these passages into a plenum chamber between the D-duct and a baffle and then is discharged tangentially into the inlet stream via several discharge ports (usually six). Heat and airflow requirements for anti-icing this engine inlet were determined at a number of different flight conditions. Since the calculations show the greatest air flow demand at the 15,000 foot (4572 m) cruise altitude, the system was designed to meet the heat requirements at this condition. The system will provide complete evaporation of the impinging water droplets in maximum continuous icing for all flight conditions except descent (due to lower bleed temperatures). The heated area will be maintained above 35°F (2°C) (running wet) for descent. In maximum intermittent icing, the heated surface will be running wet for all flight conditions. The amount of refreeze is not significant for these encounters because of the short duration of exposure. Calculated values of both actual and required heat release are shown in figure 5-35 for comparison. Figure 5-36 shows calculated values for both actual and required air flows.

The calculation of water catch, impingement limits, heat release, and air flow requirements for a turbojet engine inlet lip is described in Chapter V. Knowing the bleed air pressure available at the different flight conditions, the actual air flows can be calculated by performing a pressure drop analysis of the system. The heat release can then be found knowing the actual air flows.

For other aircraft with engines having a shorter distance from the inlet lip to the compressor face, it may be feasible to provide running wet protection for the entire area aft to the compressor face so that runback does not build up and shed into the engine. This would not be practical for the typical transport aircraft because the compressor face of its engines is on the order of 4 feet (1 m) or greater aft of the inlet lip.

5.3.4 Turbofan Components

5.3.4.1 Typical System Description

The typical hot air system bleeds off a portion of the relatively high temperature and high pressure air in the compressor gas path and uses that air to heat the component in question. The anti-icing heat available is dependent upon bleed location, ambient temperature and pressure, flight Mach number, and engine power level. The necessary elements of such a system are a bleed port at the proper compressor stage, piping or passages to transport the air between bleed port and the component, convective heat transfer passages within the component, and a location to expel the air once it has done the job of anti-icing (usually back into a low pressure region of the compressor gas path). A flow metering restriction should also be provided, whether it be an orifice or the anti-iced component itself. Such a system is an integral part of the powerplant.

Non-rotating parts such as inlet guide vanes and stators usually incorporate an external piping bleed system. The airfoil may be of single pass or multiple pass internal flow configuration, as shown in figures 5-37 and 5-38. Stiffening strips and vibration damping materials are fairly common within inlet guide vanes. An on-off valve is a standard feature so that when anti-icing is not needed the

bleed flow can be turned off. On engines with inlet guide vanes, generally the same air anti-ices both the vanes and the non-rotating spinner using a series flow system. The anti-icing hot air passes first through the vanes and then the spinner.

Anti-icing of a rotating spinner would require that the bleed air must get "on board" rotating parts internal to the engine. The most practical way to route bleed air is from the gas path through the shaft which drives the fan and spinner. The convective passages for the spinner may be double wall configuration with a narrow passage height. Such a system is usually "always on" because of the obvious difficulty with incorporating a valve in the rotating hardware.

Typical external and internal bleed anti-icing flow routing paths are shown in figure 5-39 reprinted from reference 5-7.

5.3.4.2 System Design (Compressor Bleed)

If the analyses described in Chapter V indicate that an anti-icing system is required or desirable for protection against a potential icing problem, the recommended design philosophy is to provide enough heating to maintain a running wet surface of 35°F (2°C) or greater, at the design point condition recommended below using the thermodynamic heat balance equations described in Chapter V. The design analysis should be based upon the FAR Part 25 intermittent maximum (cumuliform) cloud criteria, from which the following single design point is recommended as a minimum requirement.

Power Setting	Minimum Holding
Ambient Air Temperature	-4°F (-20°C)
Cloud LWC	1.7 g/m^3
Mean Volumetric Diameter	20 microns
Minimum Surface Temperature	35°F (2°C)

A guide vane or stator anti-icing system designed to this condition usually has the bleed source located at a mid-stage of the high pressure compressor and, in general, the vanes will accumulate some ice during lower power idle operation when there is insufficient heat available from the bleed air to anti-ice them. Ice that accumulates on inlet guide vanes during the relatively short duration of idle descent is not necessarily detrimental, although engine operability and tolerance to such ice accumulation must be thoroughly verified for both the short duration idle descent and for ground idle operation. Icing tests and analysis may show the need for powered-up engines during descent in severe icing conditions and for periodic power increases during longer duration ground operation as discussed in detail in references 5-6 and 5-7.

Some engine designers have found the need for a more complex anti-icing system which will provide higher pressure and temperature compressor discharge air to the anti-iced parts at lower power operation to alleviate the problem of insufficient heat from the low or mid-pressure compressor. This type of system incorporates a switching valve to go to higher pressure and temperature compressor air as required.

The general procedure for the final design of a hot-air anti-icing system involves: (1) using the design point above to estimate the bleed flow/temperature required to achieve a 35°F (2°C) or greater surface, (2) using the resultant bleed flow/temperature to select the required compressor bleed stage location, (3) sizing the pipes and bleed ports to get the required bleed flow at the selected temperature, and (4) performing the thermal design of the anti-iced part.

Preliminary calculations based upon one-dimensional heat transfer may be used to get a rough idea of the bleed flow and temperature required. The final detailed heat transfer analysis can be accomplished via a three-dimensional finite difference heat transfer computer analysis, using the one-dimensional result as a starting internal boundary condition. Fine adjustments to bleed flow can then be made, based upon this analysis.

5.3.4.3 Selection of Compressor Bleed Stage - Considerations

Details of a preliminary calculation procedure for determining the optimum compressor bleed location are given at the end of this section, using a typical inlet guide vane as an example. Prior to beginning the calculations, the designer should note several aspects of the selection process.

Fixing both the design point condition and engine geometry establishes the airfoil external heat load, hence, the anti-icing heat required. The internal (hot side) heat transfer coefficient is a function of the bleed flow and the vane passage geometry; therefore, one should have a rough knowledge of the minimum cross-sectional flow area (A_{flow}), hydraulic diameter (D_h), passage surface distance (S), and passage width (b), as indicated on figure 5-40. Since internal heat transfer coefficient depends upon flow, both the bleed air temperature needed to meet the required heating and the source pressure needed to deliver that flow quantity will change for various assumed values of bleed flow. Parametric curves similar to those of Figure 5-41 (reference 5-7) can be plotted to compare bleed temperature and pressure requirements against availability throughout the compressor stages.

It should be noted that the further forward in the compressor the bleed location is selected, the more the source air temperature and pressure will decrease, and the greater will be the bleed flow required. As the selected bleed air location continues to move forward in the compressor, a point will finally be reached where the stage pressure available is inadequate for delivering the required bleed flow. In addition to the above, other practical considerations are usually involved in selecting the bleed location. The hot air system must be configured to meet primary requirements for structural integrity, engine performance, cost, compressor aerodynamics, and airframe bleed requirements. Typically, the

CHAPTER III - ICE PROTECTION METHODS

CONTENTS

SECTION 6.0 SYSTEM SELECTION

CONTENTS (CONTINUED)

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SYMBOLS AND ABBREVIATIONS

Symbol	Description
APU	Auxiliary Power Unit
BTU	British Thermal Unit
°C	Degrees Celsius
cm	Centimeter
EIDI	Electro-Impulse De-Icing
EMI	Electromagnetic Interference
°F	Degrees Fahrenheit
FPD	Freezing point depressant
ft	Feet or foot
gpm	Gallons per minute
HP	Horsepower
I/P	Ice Protection
kg	Kilogram
kN	Kilonewton
kw	Kilowatt
lbf	Pounds force
lbm	Pounds mass
lbs	Pounds
m	Meter
mm	Millimeter
psig	Pounds per square inch gauge (pressure)
°R	Degrees Rankine
scfm	Specific cubic feet per minute
VCK	Variable Camber Krueger Flap
w	Watts

GLOSSARY

<u>bleed air</u> - A relatively small amount of air diverted to an auxiliary use.

<u>liquid water content (LWC)</u> - The total mass of water contained in all the liquid cloud droplets within a unit volume of cloud. Units of LWC are usually grams of water per cubic meter of air (g/m^3).

<u>median volumetric diameter (MVD)</u> - The droplet diameter which divides the total water volume present in the droplet distribution in half; i.e., half the water volume will be in larger drops and half the volume in smaller drops. The value is obtained by actual drop size measurements.

<u>micron (μm)</u> - One millionth of a meter.

<u>silver pneumatic de-icer</u> - A pneumatic de-icer which has low ice adhesive materials blended into its surface, giving the surface a silver color.

<u>stagnation point</u> - The point on a surface where the local free stream velocity is zero. It is also the point of maximum collection efficiency.

III.6.3 COMPONENT CANDIDATE SYSTEMS

6.3.1 Airfoils and Leading Edge Devices

The chordwise extent of the ice formation on an airfoil is basically a function of the ambient conditions and the airspeed. Some characteristics of the airfoil that will also affect the icing impingement limits and chordwise coverage area are:

a. Type and size of airfoil
b. Leading edge radius
c. Thickness ratio
d. Leading edge sweep angle
e. Angle of attack
f. Type and deflection of leading edge device

When selecting an icing protection system for an airfoil surface, all of the above items should be considered. Aerodynamic smoothness of the ice protection system in normal dry air flight conditions may be another important factor.

Some airfoils that have large leading edge radii may not shed ice well under all flight and ambient conditions.

The hot gas thermal systems are used on the wing of most transport category aircraft. The pneumatic boot de-ice system is used on the empennage of transport aircraft and the majority of light aircraft for wing and empennage surface ice protection. Some light aircraft have used the electro-thermal (on empennage) and the fluid injection systems with good results. Current rotorcraft use electro-thermal systems. See Chapter V for compliance requirements. Application experience with electro-impulse and pneumatic-impulse is limited as of this writing (1993). The electro-expulsive and eddy-current repulsive systems are even newer with a consequent lack of application experience.

6.3.1.1 Fixed Leading Edge

Light Aircraft (FAR 23)

The dominant factors in the selection of an ice protection system for single engine aircraft (figure 6-2) are probably cost, weight, and power available. Ice protection is usually not provided for these aircraft. When it is, the system requirements analysis should be made in the same manner as for light twin-engine aircraft (figure 6-3).

A summary of leading edge ice protection system attributes, weight, and power requirements for a typical single-engine aircraft are presented in tables 6-1 and 6-2. Values are shown for different types of systems considered to be the most desirable for this type of aircraft.

A summary of leading edge ice protection system attributes, weight, and power requirements for a typical light twin-engine aircraft (figure 6-3) are presented in tables 6-3 and 6-4. Values are shown for the types of systems considered to be the most desirable for this type of aircraft.

Transport Category Aircraft (FAR 25)

In the selection of ice protection for the wing leading edge of a typical light jet (figure 6-4), the requirements are found in the same manner as for the reciprocating twin-engine light plane. A summary of system attributes, weights, and power extraction for several different systems is presented in tables 6-5 and 6-6.

The most desirable systems for the reciprocating twin-engine airplane will not necessarily be the most desirable for the jet. More consideration may be given to hot air anti-icing systems for the wing and tail of the jet because of the availability of engine bleed air. Also, the jet aircraft may have more power available for electrical protection systems. Requirements must always be determined for ice protection of turbine engine inlets. If aft-mounted engines are used, special consideration must be given to the potential problem of ice shedding from sections of the inboard wing. The use of certain silver pneumatic de-icers, which allow thinner ice accumulations to be shed, in combination with an ice detector system, has proven successful in two large turboprop applications.

The selection process for a large multi-engine transport (figure 6-5) is much different than for the business jet because it will, in general, have sufficient power available so that several methods of ice protection become feasible. A summary of system attributes, weights, and power requirements for a typical large jet transport aircraft is shown in tables 6-7 and 6-8.

The dominant factor that affects the selection of an ice protection system for jet aircraft is probably the bleed air supply provided by the engines. Depending on the amount of bleed air available and that required for ice protection, the system selection can be divided into two categories.

a. Sufficient Supply of Bleed Air for Anti-Icing

For high performance lifting surfaces used on commercial transport aircraft, priority may be given to the hot-air anti-icing system which has been proven over the years to be a reliable and effective system. However, the projected shift to unducted fan engines and ultra-high bypass engines forces a trend toward de-icing rather than anti-icing. These types of engines, much like the turboprop engines, may not have enough compressor bleed air for anti-icing (see Section III.5.8.2).

b. Bleed Air Not Available or Insufficient for Anti-Icing

A detailed trade-off study must be conducted for the following non-bleed or low bleed alternatives:

- pneumatic boot de-icing
- pneumatic impulse de-icing
- fluid anti-icing or de-icing
- electro-impulse de-icing
- electro-expulsive de-icing
- eddy current repulsive de-icing
- electro-thermal anti- or de-icing
- hot-air de-icing (bleed air)
- hot-air anti-icing or de-icing (combustion heater)

The advantages and disadvantages of each system are shown in table 6-9. Major considerations recommended for the trade-off study are provided as follows:

a. Pneumatic boot de-icing
   - aero effects due to boots (both inflated and deflated position); can be evaluated using tube profile data provided by boot manufacturer
   - aero effects due to required ice buildup between de-icing cycles
   - aero effects due to auto-inflation (failed suction system)
   - service life of boot (may be function of hangaring practice)
   - ice ingestion into tail-mounted engines and/or propellers; work together with engine and propeller manufacturers - weight of the system
   - owner's acceptance for maintenance procedures and external appearance
   - maintenance costs
   - non-recurring costs

b. Pneumatic impulse de-icing
   - aero effects due to ice build-up between cycles
   - aero effects due to residual ice
   - ice ingestion by tail mounted engines and/or propellers
   - fatigue effects on some structural components should be considered
   - weight of system
   - noise level during pneumatic discharge
   - maintenance costs
   - non-recurring costs

c. Fluid anti-ice or de-icing
   - weight of anti-icing fluid required based on stagnation point travel, specific fluid requirement to provide anti-icing per unit area at design icing condition, and the amount of ice protection required.
   - if fluid requirement for anti-icing is too high, how about de-icing or cyclic de-icing?
   - aero effects due to ice buildup between cycles (de-icing mode)
   - ice ingestion by tail-mounted engines and/or propellers (de-icing mode)
   - fluid ingestion by tail-mounted engines
   - weight of the system
   - fluid costs and availability
   - maintenance costs
   - non-recurring costs
   - owner's acceptance (washing of aircraft usually needed after use)
   - benefits resulting from aerodynamically cleaner surfaces (less insects, dirt; protection of trailing edge)
   - environmental considerations

d. Electro-mechanical de-icing (electro-impulse, electro-expulsive, eddy current repulsion)
   - aero effects due to ice buildup between cycles
   - aero effects due to residual ice
   - ice ingestion by tail-mounted engines and/or propellers
   - fatigue effects on structural components

- weight of the system
- weight addition due to extra electric generating capacity (if required)
- electro-magnetic interference (if any)
- noise level during coil discharge
- maintenance costs
- non-recurring costs

e. Electro-thermal de-icing or anti-icing
- aero effects due to ice buildup between cycles
- ice ingestion by tail-mounted engines and/or propellers
- additional electrical generating capacity
- weight of the system
- weight addition due to extra electric generating capacity (if required)
- fuel cost for system operation
- maintenance costs
- non-recurring costs

f. Hot-air anti-icing or de-icing (bleed air)
- weight of the system
- effects on engine performance due to bleed air extraction
- fuel cost and weight penalty for system operation
- aero effects due to ice buildup between cycles (de-icing mode)
- ice ingestion by tail-mounted engines and/or propellers (de-icing mode)
- maintenance costs
- non-recurring costs

g. Hot-gas anti-icing or de-icing (combustion heaters)
- air compression ratio to determine size and weight of compressor and motor
- weight of the system
- weight addition due to extra electric generating capacity (for electric-driven motor)
- fuel cost and weight penalty for system operation
- aero effects due to ice buildup between cycles (de-icing mode)
- ice ingestion into tail-mounted engines and/or propellers (de-icing mode)
- maintenance costs
- non-recurring costs

6.3.1.2 Leading Edge Slots, Slats, and Krueger Flaps

The discussion of fixed leading edge system selection (Section III.6.3.1.1) is applicable to slot and slat configurations; with the exception that slats require a means of transmitting the heated air, freezing point depressant (FPD), or electrical power from the fixed wing to the moveable slat. A telescoping duct that is capable of rotating and extending has been used for hot air systems. Swivel fitting and flexible hoses have been used for pneumatic de-icers and liquid FPD transmission. In all cases, the means of transmission presents an added degree of complexity and potential failure that must be considered during the trade-off studies used as a basis of system selection.

Since the Krueger flap retracts into the lower airfoil section, many applications have not required ice protection. When ice protection is required, any of the concepts discussed in Section III.6.3.1.1 may be considered. In addition, a variable camber Krueger (VCK) has been used that provides automatic de-icing. In this design, the linkage that extends the VCK also alters the camber to increase lift. During retraction after ice has accumulated, flexing the VCK debonds and shatters the ice cap allowing aerodynamic forces to remove the ice. This concept has the normal penalties of de-icing systems as discussed in Section III.6.3.1.1. In addition, the flexing may not debond the leading edge ice sufficiently to allow removal by aerodynamic forces.

6.3.2 Control Surface Balance Horns

Control surfaces having leading edges which can collect ice must be protected. Often the leading edge is in the shadow of the fixed surface and does not collect ice; e.g., ailerons and split flaps. But if the leading edge rotates out of the wing plane, it will leave the protection of the upstream airfoil and is in danger of collecting ice in such quantity as to jam the control surface in the extended position. Examples are rudder or elevator balance horns and Fowler flaps. Decreasing the gap between the fixed and moving surface helps to decrease the danger, but active ice protection for the leading edge may be required.

Ice protection system candidates for this application are limited due to the small size and hinge-type supports. An electrical method is probably best, either electro-thermal or electro-mechanical.

Icing tunnel or flight tests may be necessary to determine the extent of the problem and decide whether upstream shielding or de-icing is needed.

6.3.3 Windshields

Aircraft with ice protection commonly use electrically heated laminated windshields. Typically, these systems consist of vacuum deposited metallic coatings applied to a glass surface equipped with contacts and temperature sensing devices. The coated glass plate is then laminated to other plates of glass and plastic depending on the specific windshield design (references 6-1 and 6-2).

Another electrical system design uses very fine wires laminated in the plates of glass and plastic. Electrical power is supplied to the wires through two bus bars at the ends of the windshield.

Both of these systems require a controller to regulate the power applied to the heating element to ensure adequate anti-icing performance and to prevent overheating of the windshield. These systems may also provide defogging of the windshield if designed so the internal windshield surface temperature is above the cockpit air dew point.

A fluid de-icing system has been used on some aircraft for windshield protection. This system uses a freezing point depressant, such as an ethylene glycol/water mixture, which is sprayed on the windshield. This produces a slush which is then blown off the windshield by aerodynamic forces.

The principal disadvantage is the residue left by the fluid and the quantity of fluid required for adequate protection.

On aircraft powered by turbine engines, with an abundant supply of bleed air, an external hot air windshield anti-ice system may be considered. This system may also be utilized for rain removal. A major design problem is maintaining the correct air flow and temperature requirements for icing without overheating the windshield. The air flow and temperature requirements for anti-icing may be less than for rain removal.

Although hot air anti-iced windshields have been used in the past, all current commercial transport aircraft use electrically heated anti-icing systems. The selection of detailed design features of each windshield is influenced by the unique features of the particular aircraft and must consider the integrated design and requirements of the entire windshield. Some of the paramount considerations in addition to normal structural and anti-icing concerns are: hail impact, bird strike, pressurization, environmental extremes including transient effects, rain removal, fogging, visibility, materials properties, lightning, static electrical charge, electrical system design, plus numerous considerations based on in-service experience. Obviously, the design of a windshield must be a closely coordinated activity of a team including structural, electrical, and environmental engineering; specialists in materials, lightning strikes, and manufacturing processes; and the specific vendor(s).

Care must be exercised in the design of the maximum heating capabilities of any windshield anti-icing (de-icing) system. Excessive temperature applied to the windshield could cause fogging of the plastic laminates, crazing, delamination and/or distorted visibility.

6.3.4 Engine Inlet Lips and Components

Icing protection for engine inlets can be hot air, electrical, pneumatic, pneumatic impulse or electro-mechanical. On turbine engine aircraft, the inlets are usually heated with bleed air. They are frequently designed to be evaporative under continuous maximum icing conditions and running wet under intermittent maximum conditions. Low power electro-mechanical systems may be used. Also, electro-thermal protection may be used if adequate electrical power is available.

Extra precautions should be taken to prevent engine damage, by ice shedding into the inlet or by possible runback and refreeze that could be shed into the engine. Aerodynamic surfaces ahead of the engine inlet should be considered as extensions of the engine inlet lips and ice protection should be provided accordingly. The ice particle shed-size can be minimized with electro-mechanical and pneumatic impulse de-icing systems if the power required to anti-ice is not available. The pneumatic de-icer can be used with turbine engines which have a by-pass feature which allows the removed ice particles to be discharged through the by-pass duct.

The concept generally selected for turbojet inlet ice protection has been hot air anti-icing. The main considerations in making this decision have been availability of bleed air, high-reliability of the installation, and the excellent anti-icing performance of bleed air. Recently, due to the high cost of fuel and the limited availability of excess bleed air from the newer high-bypass ratio engines, more consideration is being given to electro-mechanical and pneumatic-impulse de-icing, and fluid anti-icing. Electro-thermal ice protection is feasible for the smallest turboprop inlets, but is generally impractical on turbofan engine inlet lips due to the high power consumption requirements.

Both de-icing and anti-icing systems may be considered for engine inlets. The primary concern regarding ice buildup for a de-icing system is the engine limitation on ice ingestion. The effect of ice buildup on inlet flow aerodynamics is generally less critical than on wings except in the inlet throat region.

The selection of the ice protection system for an engine inlet should be based on a cost-of-ownership analysis and weighted by factors such as development risk and the general preference for anti-icing as opposed to de-icing. Some of the factors to be considered in any trade-off study are summarized in table 6-10.

For a more complete treatment of ice protection for engine components, the reader is directed to reference 6-3.

6.3.5 Turbofan Components

The design engineer who is faced with configuring an engine anti-icing system must be aware of the various types of systems from which to choose, their effectiveness, relative complexity and reliability, and other practical limitations. Typical systems for consideration for turbofan engines are electrical and compressor hot air bleed. The design engineer should also be aware of hot scavenge oil systems for stators which provide anti-icing protection in some turboshaft engines, as described in reference 6-4.

6.3.5.1 Electrical Systems for Turbofans

A typical electrical anti-icing system which embeds resistance heaters within the component to be protected obtains its energy from the airframe supplied generator. Such a system would not be an integral part of the powerplant and sizing of the generator would be affected by the engine anti-icing requirements. Electrical heating of rotating components, such as a spinner, would require the incorporation of a slip ring arrangement which would probably present a reliability problem. A particular merit of an electrical system would be the ability to embed the heat source close to the leading edge of a very thin stator where routing of hot air to achieve sufficient anti-icing effectiveness would be difficult. All aspects considered, it appears that electrical anti-icing systems for turbine engines are not generally accepted by the industry.

6.3.5.2 Hot Air Systems for Turbofans

Thermal systems may vary in the level of protection they provide, ranging from complete evaporation of all water that impinges, to allowing the body surface to run wet (with water runback) at a preselected temperature above freezing. Furthermore, a periodic de-icing system is also a feasible alternative in some cases. In general, a system that achieves a high degree of evaporation on engine components is also one that requires an uneconomically high heat input. Therefore, the accepted design philosophy is to provide a running wet surface in icing conditions. The designer should also investigate closely the hot air flow requirements for an anti-icing system, which could be a substantial

amount of the total engine air flow, with an adverse effect on engine performance. A hot air anti-icing system is the major type of system used on large transport aircraft engines of today for protecting inlet guide vanes and nose cones. Typical hot air engine anti-icing systems are described in Section III.5.2.3.

6.3.6 Propellers, Spinners, and Nose Caps

The most popular current propeller anti-icing system used today is an electro-thermal boot which is bonded to the propeller blades. The heating element usually consists of a stainless steel ribbon embedded in the boot. This heating element is designed to provide more heat at the root of the propeller blade, where the ice buildup is the heaviest and progressively less heat outboard, where centrifugal force reduces buildup problems.

Total boot coverage is approximately 15% of the chord on the suction surface and approximately 30% of the pressure surface. The power to the boots is supplied through slip rings at the propeller hub. Spanwise extent of the heated surface is usually about 30% of blade length.

The propeller thermal anti-icing requirements will vary depending upon propeller diameter, rotational speed, and aircraft forward velocity.

Another propeller anti-icing system is the fluid system, which uses an ethylene glycol/water mixture pumped to the propeller hub and dispersed to the blades through slinger rings. The system is limited by the quantity of the fluid.

De-icing by electro-thermal boots is also used, with the electric power sequenced from one blade segment to the other. Cyclic heating must be sequenced so as to avoid asymmetric shedding. For an even number of blades, opposing blades are heated simultaneously moving from large radius to small radius positions. For an odd number of blades, all blades must be deiced at the same radial sections simultaneously. Outer positions are deiced first to avoid acting as a dam for the debonded inner ice which is being pushed outward by centrifugal forces. Electro-mechanical de-icing systems may also be considered using similar logic.

6.3.7 Helicopter Rotors, Hubs, and Droop Stops

6.3.7.1 System Selection for Rotorcraft

The selection of appropriate ice protection systems for rotorcraft requires a knowledge of mission requirements and constraints. Preceding sections have presented the attributes of pneumatic boot, pneumatic-impulse, electro-thermal, systems, fluid injection, electro-mechanical, and hot air systems. Those sections provide information that describe potential applications, many of which are common to both airplanes and rotorcraft. Sections III.6.3.1, III.6.3.3, III.6.3.4, III.6.3.8, III.6.3.9 and III.6.3.10 are directly applicable to rotorcraft and need not be discussed further in this section. The following sections deal only with the application of de-icing and anti-icing systems to problems that are unique to rotorcraft.

6.3.7.2 Distinctive Rotorcraft Icing Problems

While the rotor has many of the attributes of the propeller, the flight environment and larger diameter tend to set the required ice protection systems apart from propeller ice protection system design. Droplets from supercooled clouds generally freeze forward of 15% chord on the upper airfoil surface and 25% chord on the lower airfoil surface. An anti-icing system that melts ice only in this region will result in an ice formation aft of the heated region due to refreezing of melted ice. The resultant penalties in the lift, drag, and pitching moment are not acceptable. Thermal anti-icing of the whole blade surface may impose unacceptable increases in system cost, weight, and power. A fluid anti-icing system, which lowers the freezing temperature of the droplets below the ambient temperature, avoids the runback problem. However, at present no fluid rotor ice protection system has been certified. Rotor hubs are generally exposed on a helicopter, but experience has shown that the only components requiring ice protection are droop stops which tend to freeze in the "fly" position if the hinge pin is unheated. Stores support systems are unprotected and attention must then be given only to the prevention of ice shedding into engines and rotors. Engines must be qualified for flight not only in forward flight, but also in hover, rearward, and sideward flight. Although icing may not actually occur in these flight modes, the helicopter must be able to approach and land after flight in icing.

6.3.7.3 Rotors

The selection of a rotor ice protection system involves a thorough understanding of mission requirements and design constraints. The eventual solution may he possible to justify in terms of weight and cost, but other factors may enter into the decision-making process. The current level of maturity of a concept may be important to a designer, but other goals may require investigation of advanced systems which, as of this writing, have not been applied to modern helicopters. Advances in ice detection and accretion measurement systems may also become a part of the de-ice system selection process.

The electro-thermal system, the only de-ice system now in production, is in use on the Aerospatiale AS332 Super Puma, Sikorsky UH-60A BLACK HAWK, and SH-60B SEAHAWK. The CHSS-2 (Canadian Armed Forces Sikorsky S-61A) electro-thermal system was previously in production. Several helicopter models are also in various stages of de-ice system qualification (Boeing Vertol Model 234 and HC-MK1, Bell 214ST and 412, Bell/Boeing V-22, McDonnell Douglas AH-64A, and Sikorsky S-76B) and each of these rotorcraft uses electro-thermal rotor de-ice systems. Electro-thermal systems have been tested on Aerospatiale SA330, Bell UH-1H, MBB BO-105, Sikorsky H-34, SH-3, and S-76A, and Westland Wessex 5 helicopters. While this type of system would appear to he the overwhelming choice of designers, this is essentially by default. The level of technology available in the 1970s precluded the use of fluid, vibratory, and pneumatic boot systems and development had not started on the pneumatic-impulse de-icing system (PIDI). Work on the electro-impulse de-icing system (EIDI) prior to 1980 was conducted in the Soviet Union, but development in the United States

did not start in earnest until 1982. As described in greater detail in the preceding sections, research is continuing on alternatives to electro-thermal systems, and advances in the state-of-the-art for each system are expected. Flight tests were conducted in the early 1960s on a UH-1 equipped with a fluid anti-icing system and recent NASA airfoil tests studied fluid protection systems in both anti-icing and de-icing modes. The UH-1H has been tested with a pneumatic rotor ice protection system. A main rotor hot air anti-icing system was tested on a Sikorsky S-51 (H-5) helicopter. This used a 200,000 BTU/hr (70 HP) heater with sufficient hot air introduced into the blade spar to keep the leading edge free of ice. The system did keep the leading edges free of ice at less than critical conditions, but the moisture froze on the unheated blade surface. The system was deemed impractical for this and other reasons. Ice phobic tests on an AH-56 helicopter shows no benefits. Shear stresses must be as low as 1 psi to have an effective system. Tail rotor ice protection systems are derived using techniques applicable to both rotors and propellers. The following protection limits are recommended for main and tail rotors:

		Main Rotor	Tail Rotor
Chordwise Coverage -	Upper Surface	0 to 15%	0 to 13%
	Lower Surface	0 to 25%	0 to 13%
Spanwise Coverage -		20% to 99%	25% to 70%

The main part of the ice formation generally forms forward of 8% chord. The following material summarizes the attributes of rotor de-icing and anti-icing systems and is intended to serve as a guide for system selection. Tabular summaries are given in tables 6-11, 6-12, and 6-13 for the helicopter shown in figure 6-6.

Blade construction may dictate the type of system to be used. A typical rotor blade leading edge is solid, containing counterweights and filler material. Since fluid and EIDI systems are added to an airfoil section internally, retrofit of these systems to existing blades would be an expensive process. Therefore, these systems are currently not recommended as add-on ice protection systems. Also, at the current stage of coil development, application of EIDI to airfoils with chords less than 15 inches is not practical.

Useful life, reliability, and maintainability must be considered in the system selection. Rotor blades are designed with nickel or titanium leading edge abrasion protection to withstand the harsh environment imposed on a rotor operating in rain or near the ground. The maintainability of an erosion resistant pneumatic de-icing boot has been demonstrated during flight testing, but experiments conducted to date have not been sufficient to determine boot life during helicopter operations, or the costs associated with system maintenance. Metallic abrasion strips can be incorporated. into other designs to reduce or eliminate the impact of the ice protection system on blade erosion, but special techniques may be necessary to adapt the abrasion strip to the de-ice system requirements.

Each system involves either an electronic or mechanical union to transfer signals from the controller and power sources to the blades. Fluid systems, while mechanically simple, must be kept

III.6.4 WEIGHT, ENERGY, AND POWER COMPARISONS

Weight, energy and power requirement comparisons for ice protection system on four typical aircraft (figure 6-2 through 6-5) are presented in tables 6-1 through 6-8.

III.6.5 DISTINCTIVE SMALL AIRPLANE ICING PROBLEMS (FAA PART 23)

Aircraft in this category (reference 6-5) can be single or multi-engine aircraft with a maximum takeoff weight of 12,500 pounds or less, and with either reciprocating or turbine engines. On aircraft with reciprocating engine installations, pneumatic boot, pneumatic-impulse, electro-thermal, fluid injection, or electro-mechanical icing systems may be selected for lifting surface ice protection. On turbine engine aircraft, hot-air anti-icing systems may also be considered, due to the availability of engine bleed air as a heat source. On small aircraft, the total icing system weight and cost are major factors in selecting a satisfactory system.

Ice protection may be more critical for small airplanes than for larger aircraft. Overflying an icing cloud is often impossible due to lower service ceilings. Their limited range makes weather avoidance or using alternate landing sites more difficult. There are usually more non-protected components, such as struts, non-retracting wheels, and steps. Smaller component sizes tend to have higher collection efficiencies, thus relatively larger ice accretions occur. Engine power margins are less and excess engine power to drive accessories is limited. The single engine airplane inherently has reduced system redundancies and thus a reduced reliability. FAR23.1309 states that equipment, systems, and installations of a single engine airplane must be designed to minimize hazards to the airplane whereas those of a multi-engine airplane must be designed to prevent hazards in the event of a probable malfunction or failure.

III.6.6 DISTINCTIVE TRANSPORT CATEGORY AIRPLANE PROBLEMS (FAR PART 25)

Aircraft in this category can be either small, medium or large and each would have somewhat distinctive ice protection problems. Added to this is the complication that these aircraft can be powered by piston engines, turboprop engines and propellers, or jet engines (high or low bypass ratio).

A distinctive advantage of these aircraft lies in the availability of bleed air and relatively large amounts of power for ice protection. Thus all forms of ice protection can be supported. The use of more than one type is likely because this flexibility allows the designers to install the optimum system for each component that is to be protected. With the advent of reduced bleed-air engines, the use of low power de-icing systems such as pneumatic, pneumatic-impulse, or electro-mechanical systems should begin to play a more major role on this class of aircraft.

An ice protection area that is perhaps unique to this category of aircraft is wing leading edge devices. The possible need for radome ice protection is a consideration for aircraft in this category which need not be addressed for some Part 23 aircraft.

III.6.7 DISTINCTIVE ROTORCRAFT ICING PROBLEMS (FAR PART 27/29)

The protection of helicopter components presents some unique problems. Early designers thought that vibration and centrifugal forces would prevent appreciable ice accretion on helicopter rotors. This is not the case - ice does accrete and the increased drag causes an increase in the power requirements of the helicopter. On small helicopters, the increase in airfoil drag may be sufficient to force the aircraft to land, while on some large helicopters the power increments may be acceptable. When self-shedding does occur, it is usually asymmetrical, and this causes an imbalance that may result in severe vibration.

Ice accretion in relatively small quantities may create severe hazards to a helicopter. These hazards may be from excessive vibration caused by asymmetrical self-shedding of ice from the main and tail rotors, from ice impact damage when self-shedding occurs, or from the increase in airfoil drag when self-shedding does not occur.

Another area of a helicopter that may require ice protection is the rotor head mechanism. The rotor head mechanism is not likely to freeze up during flight or malfunction if the mechanism is actuated frequently. But, if icing is a hazard to the mechanism, then a simple windscreen or some de-icing system may be necessary.

Propulsion system ice protection may consist of inlet screen anti-icing and turbine engine inlet anti-icing. On one flight test program involving a light, turbine-powered helicopter, the inlet screen iced in a waffle pattern and caused approximately 15 percent air blockage. The total open area of the inlet screen was twice the engine inlet area; consequently, the ice blockage did not affect the engine performance. This area ratio may not always occur, and icing on an inlet screen may cause sufficient blockage to affect engine performance. Other inlet screens for compartment cooling, carburetor inlets, etc., may also require ice protection.

III.6.8 REFERENCES

6-1 Lawrence, James H., "Guidelines for the Design of Aircraft Windshields/ Canopy Systems," AFWAL-TR-80-3003, February 1980.

6-2 Hassard, Richard S., "Plastics for Aerospace Vehicles, Part II, Transparent Glazing Materials," MIL-HDBK-17A, Part II, January 1973.

6-3 Pfeifer, G.O. and Maier, G.P., "Engineering Summary of Powerplant Icing Technical Data," FAA Report RD-77-76, July 1977.

6-4 Duffy, R.J. and Shattuck, B.F., "Integral Engine Inlet Particle Separator, Volume II - Design Guide," Report Number USAAMRDL-TR-75-31B, August 1975.

6-5 "Aircraft Ice Protection," FAA Advisory Circular 20-73, April 1971.

6-6 Bowden, D.T., Gensemer, A.E. and Speen, C.A., "Engineering Summary of Airframe Icing Technical Data," FAA Technical Report ADS-4, March, 1964.

TABLE 6-1. AIRCRAFT ICE PROTECTION SYSTEM ATTRIBUTES
FAR 23 - SMALL SINGLE ENGINE AIRCRAFT

AIRCRAFT COMPONENT	SYSTEM TYPE	WEIGHT lbs.	POWER KW	BLEED AIR lb/min	POTENTIAL FOR PROBLEMS ? PARTIAL SHED	RUN-BACK	FATIGUE
Wing and Tail	Pneumatic De-Icing	25		Negl.	Low	None	None
Windshield	Elec. Anti-Icing	8	1.4		Low	Low	None
Propeller	Elec. De-Icing	5	0.6		Some	None	None
	Generator	10					
Totals		48	2.0				
Wing and Tail	Elec. Impulse De-Ice	50	0.3	None	Low	Some	Yes
Windshield	Elec. Anti-Icing	8	1.4		Low	Low	None
Propeller	Elec. De-Icing	5	0.6		Some	None	None
	Generator	10					
Totals		73	2.3				
Wing and Tail	Fluid Anti-Icing	40	Negl.	None	No	Low	None
Windshield	Fluid Anti-Icing	3	Negl.		Low	Low	None
Propeller	Elec. De-Icing	2	0.6		Some	None	None
	Fluid	55					
Totals		100	0.6				
Wing and Tail	Pneumatic Impulse	57	0.15		Low	None	Some
Windshield	Elec. Anti-Icing	8	1.4		Low	Low	None
Propeller	Elec. De-Icing	5	0.6		Some	None	None
	Generator	10					
Totals		80	2.15				

Notes: Electro-thermal for wings and tail exceeds power available.
Hot bleed air usually not available on this class aircraft.
All numbers are approximate.

TABLE 6-2. WEIGHT SUMMARY - WING AND TAIL SYSTEMS
FAR 23 - SMALL SINGLE ENGINE AIRCRAFT

SYSTEM ----------->	PNEUMATIC BOOTS	EIDI	FLUID PROTECTION	PNEUMATIC IMPULSE
System Type	De-Icing	De-Icing	Anti or De-Icing	De-Icing
Alternators	0	0	0	0
Controls	3	16	1.5	3
Wiring	0	12	1	0
Distributors	2	0	3	15
Coils and Mounts	0	22	0	0
Surface Modifications	20	0	20	24
Fluid	0	0	50	0
Pumps, Tanks, Trapped Fluid, Fluid Reservoir, Compressor	0	0	14.5	15
Weight Totals (pounds approx.)	30	50	90	57

TABLE 6-3. AIRCRAFT ICE PROTECTION SYSTEM ATTRIBUTES
FAR 23 - SMALL TWIN ENGINE AIRCRAFT

AIRCRAFT COMPONENT	SYSTEM TYPE	WEIGHT lbs.	POWER KW	BLEED AIR lb/min	POTENTIAL FOR PROBLEMS ? PARTIAL SHED	RUN-BACK	FATIGUE
Wing and Tail	Pneumatic De-Icing	28		Negl.	Low	None	None
Windshield	Elec. Anti-Icing	10	1.5		Low	Low	None
Propeller	Elec. De-Icing	10	0.8		Some	None	None
	Generator	10					
Totals		58	2.3				
Wing and Tail	Elec. Impulse De-Ice	63	0.5	None	Low	None	Yes
Windshield	Elec. Anti-Icing	10	1.5		Low	Low	None
Propeller	Elec. De-Icing	10	0.8		Some	None	None
	Generator	12					
Totals		95	2.8				
Wing and Tail	Fluid Anti-Icing	40		None	None	Low	None
Windshield	Fluid Anti-Icing	4			Low	Low	None
Propeller	Elec. De-Icing	10	1.2		Low	Some	None
	Fluid	65					
Totals		119	1.2				
Wing and Tail	Pneumatic Impulse	84	0.22		Low	None	Some
Windshield	Elec. Anti-Icing	10	1.5		Low	Low	None
Propeller	Elec. De-Icing	10	0.8		Some	None	None
	Generator	12					
Totals		116	2.52				

Note: All numbers are approximate.

TABLE 6-4. WEIGHT SUMMARY - WING AND TAIL SYSTEMS
FAR 23 - SMALL TWIN ENGINE AIRCRAFT

SYSTEM ---------->	PNEUMATIC BOOTS	EIDI	FLUID PROTECTION	PNEUMATIC IMPULSE
System Type	De-Icing	De-Icing	Anti or De-Icing	De-Icing
Alternators	0	0	0	0
Controls	4	17	3	4
Wiring	0	15	2	0
Distributors	4	0	4	23
Coils and Mounts	0	31	0	0
Surface Modifications	30	0	23	36
Fluid	0	0	60	0
Pumps, Tanks, Trapped Fluid,	0	0	14	21
Weight Totals (pounds approx.)	38	63	106	84

TABLE 6-5. AIRCRAFT ICE PROTECTION SYSTEM ATTRIBUTES
FAR 25 - BUSINESS JET AIRCRAFT

AIRCRAFT COMPONENT	SYSTEM TYPE	WEIGHT lbs.	POWER KW	BLEED AIR lb/min	POTENTIAL FOR PROBLEMS ? PARTIAL SHED	RUN-BACK	FATIGUE
Wing and Tail	Hot Air Anti-Icing	80		42[*]	Yes	Yes	None
Windshield	Elec. Anti-Icing	10	1.5		Low	Low	None
Engine Inlet	Hot Air Anti-Icing	15		13	Yes	Yes	None
	Generator	10					
Totals		115	1.5	55			
Wing and Tail	Elec. Impulse De-Ice	90	0.7		Low	None	Yes
Windshield	Elec. Anti-Icing	10	1.5		Low	Low	None
Engine Inlet	Elec. Impulse De-Ice	20	0.2		Low	None	Yes?
	Generator	10					
Totals		130	2.4				
Wing and Tail	Fluid Anti-Icing	110	0.1		None	Low	None
Windshield	Fluid Anti-Icing	10			None	Low	None
Engine Inlet	Hot Gas Anti-Icing	15		13	Yes	Yes	None
Totals		135	0.1	13			
Wing and Tail	Pneumatic De-Icing	35		Negl.	Low	None	None
Windshield	Elec. Anti-Icing	10	1.5		Low	Low	None
Engine Inlet	Elec. Anti-Icing	10	8.8		Low	Low	None
	Generator	20					
Totals		75	10.3				
Wing and Tail	Elec. De-Icing	20	11.7		Some	None	None
Windshield	Elec. Anti-Icing	10	1.5		Low	Low	None
Engine Inlet	Elec. Anti-Icing	10	8.8		Low	Low	None
	Generator	20					
Totals		60	22.0				
Wing and Tail	Pneumatic Impulse	75	0.22		Low	None	Some
Windshield	Elec. Anti-Icing	10	1.5		Low	Low	None
Engine Inlet	Pneumatic Impulse	15	0.08		Low	None	Some
Totals		100	1.8				

Notes: All numbers are approximate. [*] About 2.5% of Engine Core Airflow.

TABLE 6-6. WEIGHT SUMMARY - WING AND TAIL SYSTEMS
FAR 25 - BUSINESS JET AIRCRAFT

SYSTEM ------->	PNEUMATIC BOOTS	EIDI	HOT AIR	FLUID PROTECTION	ELECTRO-THERMAL	PNEUMATIC IMPULSE
System Type	De-Icing	De-Icing	Anti- or De-Icing	Anti or De-Icing	De-Icing	De-Icing
Alternators	0	0	0	0	10	0
Controls	5	18	9	3	3	5
Wiring	0	22	1	2	2	0
Distributors	5	0	40	4	0	19
Coils and Mounts	0	50	0	0	0	0
Surface Modifications	25	0	30	26	15	30
Fluid	0	0	0	60	0	0
Pumps, Tanks, Trapped Fluid,	0	0	0	15	0	21
Weight Totals (pounds approx.)	35	90	80	110	30	75

TABLE 6-7. AIRCRAFT ICE PROTECTION SYSTEM ATTRIBUTES
FAR 25 - LARGE TRANSPORT CATEGORY AIRCRAFT (250 PASSENGERS)

AIRCRAFT COMPONENT	SYSTEM TYPE	WEIGHT lbs.	POWER KW	BLEED AIR % CORE FLOW	POTENTIAL FOR PROBLEMS ? PARTIAL SHED	RUN-BACK	FATIGUE
Wing and Tail	Hot Air Anti-Icing[2]	190		2.5	Low	Yes	No
Windshield	Elec. Anti-Icing[1]	25	3.0		Low	Low	No
Engine Inlet	Hot Air Anti-Icing	45		0.7	Low	Yes	No
Totals		260	3.0	3.2			
Wing and Tail	Elec. Impul. De-Ice[3]	400	2.6		Low	No	Yes
Windshield	Elec. Anti-Icing	25	3.0		Low	Low	No
Engine Inlet	Elec. Impul. De-Ice	90	0.6		Some	No	Yes
Totals		515	6.2				
Wing and Tail	Fluid Anti-Icing	340	0.1		No	Low	No
Windshield	Fluid Anti-Icing	25	3.0		Low	Low	No
Engine Inlet	Hot Air Anti-Icing	45		0.7	Low	Yes	No
Totals		410	3.1	0.7			
Wing and Tail	Pneumatic De-Icing	195			Low	No	No
Windshield	Elec. Anti-Icing[1]	25	3.0		Low	Low	No
Engine Inlet	Hot Air Anti-Icing	45		0.7	Low	Yes	No
Totals		265	3.0	0.7			
Wing and Tail	Elec. De-Icing	140	60.0		Low	Low	No
Windshield	Elec. Anti-Icing[1]	25	3.0		Low	Low	No
Engine Inlet	Hot Air Anti-Icing	45		0.7	Low	Yes	No
Totals		210	63.0	0.7			
Wing and Tail	Pneumatic Impulse	362	0.22		Low	Low	Some
Windshield	Elec. Anti-Icing	25	3.0		Low	Low	No
Engine Inlet	Pneumatic Impulse	70	0.08		Low	None	Some
Totals		457	3.3				

Notes: All numbers are approximate.
1. Weight for windshield anti-icing includes extra generator capacity.
2. Excludes weight of ducts and controls shared with other bleed air uses.
3. Includes partial redundancy in the system.

TABLE 6-8. WEIGHT SUMMARY - WING AND TAIL SYSTEMS
FAR 25 - LARGE TRANSPORT CATEGORY AIRCRAFT

SYSTEM ------->	PNEUMATIC BOOTS	EIDI[1]	HOT AIR[2]	FLUID PROTECTION	ELECTRO-THERMAL	PNEUMATIC IMPULSE
System Type	De-Icing	De-Icing	Anti- or De-Icing	Anti or De-Icing	De-Icing	De-Icing
Alternators	0	0	0	0	50	0
Controls	5	100	15	3	40	10
Wiring	0	100	0	3	18	0
Distributors	15	0	135	20	0	100
Coils and Mounts	0	200	0	0	0	0
Surface Modifications	175	0	40	110	32	210
Fluid	0	0	0	150	0	0
Pumps, Tanks, Trapped Fluid,	0	0	0	54	0	42
Weight Totals (pounds approx.)	195	400	190	340	140	362

Notes:
1. System provides for partial redundancy.
2. Excludes weight of ducts and controls shared with other bleed air uses.

TABLE 6-9. ADVANTAGES AND DISADVANTAGES OF ICE PROTECTION SYSTEMS

SYSTEM	ADVANTAGES	DISADVANTAGES
Hot Air (Bleed)	Conventional method Good I/P performance (anti-icing) Easy to maintain	Reduced engine efficiency and power Typical de-icing penalties* (if operated in de-icing mode)
Hot Air (combustion heater)	Good I/P performance (anti-icing) Non-bleed method	High weight penalty Expensive to operate Typical de-icing penalties* (if operated in de-icing mode)
Pneumatic Boot	Low initial cost Light weight Low bleed-air requirement Proven - in common use Long history	Aero effects due to deicer itself Aero effects due to residual ice Typical de-icing penalties* Short service life Poor appearance
Pneumatic Impulse	Low power consumption Thin ice removal Small ice particle shed Non-bleed method Long life Aero. non-intrusive Low visual profile	Little operational experience Need to consider fatigue in design Lower but typical de-ice penalties
Fluid Protection	Good I/P performance (anti-icing) Relatively low maintenance Non-bleed method	Engine ingestion of fluid Typical de-icing penalties* (if operated in de-icing mode) High initial cost
Electro-Mechanical	Low power consumption Low maintenance Non-bleed method	Unproven for transport aircraft Possible fatigue in structures Typical de-icing penalties* Aero effects due to residual ice
Electro-Thermal	Non-bleed method	Excessive power consumption for large area of I/P coverage Typical de-icing penalties*

* Typical de-icing penalties:
Aerodynamic effects prior to each de-icing cycle due to ice buildup.
Possible ice ingestion into tail-mounted engine inlets or propellers.

TABLE 6-10. SIGNIFICANT ENGINE INLET ICE PROTECTION SELECTION CONSIDERATIONS

SYSTEM TYPE ------>	HOT AIR ANTI-ICE	HOT AIR DE-ICE	ELECTRO-MECHANICAL DE-ICE	FLUID ANTI-ICE	FLUID DE-ICE	ELECTRO-THERMAL ANTI-ICE	ELECTRO-THERMAL DE-ICE	PNEUMATIC DE-ICE	PNEUMATIC IMPULSE DE-ICE
1. Aerodynamic effects of ice cap		X	X		X		X	X	X
2. Aerodynamic effects of runback ice	X					X			
3. Ice ingestion (de-icing)		X	X		X		X	X	X
4. Ice ingestion (runback)	X					X			
5. Bleed air penalty	X	X							
6. Weight penalty	X	X	X	X	X	X	X	X	X
7. Weight for additional electrical power generation						X	X		
8. Power penalty	X					X	X		
9. Fluid costs				X	X				
10. Maintenance costs	X	X		X	X	X	X	X	X
11. Non-recurring costs	X	X	X	X	X	X	X	X	X
12. Fluid ingestion and toxicity				X	X				
13. EMI			?			X	X		
14. Noise	?	?	X						X
15. Fatigue	?	?	X						X
16. Development risk			X	X	X	X	X		X

TABLE 6-11. ROTORCRAFT MAIN ROTOR DE-ICING SYSTEM ATTRIBUTES
FAR 27/29 ROTORCRAFT (1)
SYSTEM SIZED FOR FOUR 24-FOOT MAIN ROTOR BLADES

SYSTEM	Pneumatic Boots	Electro-Thermal	EIDI	Fluid Protection	Vibratory	Ice Phobic
SYSTEM TYPE	De-Icing	Anti- or De-Icing	De-Icing	Anti- or De-Icing		
Weight, Pounds	54	84	110	192	110	34
Dry Air Parasite Drag, Ft^2	0.7	0.7	Negl	0	Negl	0
Dry Air Profile Power Delta, Hp	50 (2)	25 (2)	0	(4)	0	0
Electrical Extraction, HP	Negl	35 (3)	1	1	2	0
Bleed Air Power Loss, HP	Negl	0	0	0	0	0
Minimum Ice Thickness, In	.25	.15	.15	0	.30	.30
Average Power Rise Due to Ice, %	20	15	15	0	25	25
Partial Shed Potential	Low	Low	Low	None	High	High
Runback Potential	No	Yes	No	No	No	No
Fatigue Stress Increase	No	No	Yes	No	Yes	No

(1) All numbers are approximate and apply to the helicopter in Figure 6-6.
(2) Delta HP = 0 for in-contour installation.
(3) Approximately 60 horsepower is required for the CH-47 Chinook tandem rotor helicopter.
(4) Depends on airfoil porosity and roughness.

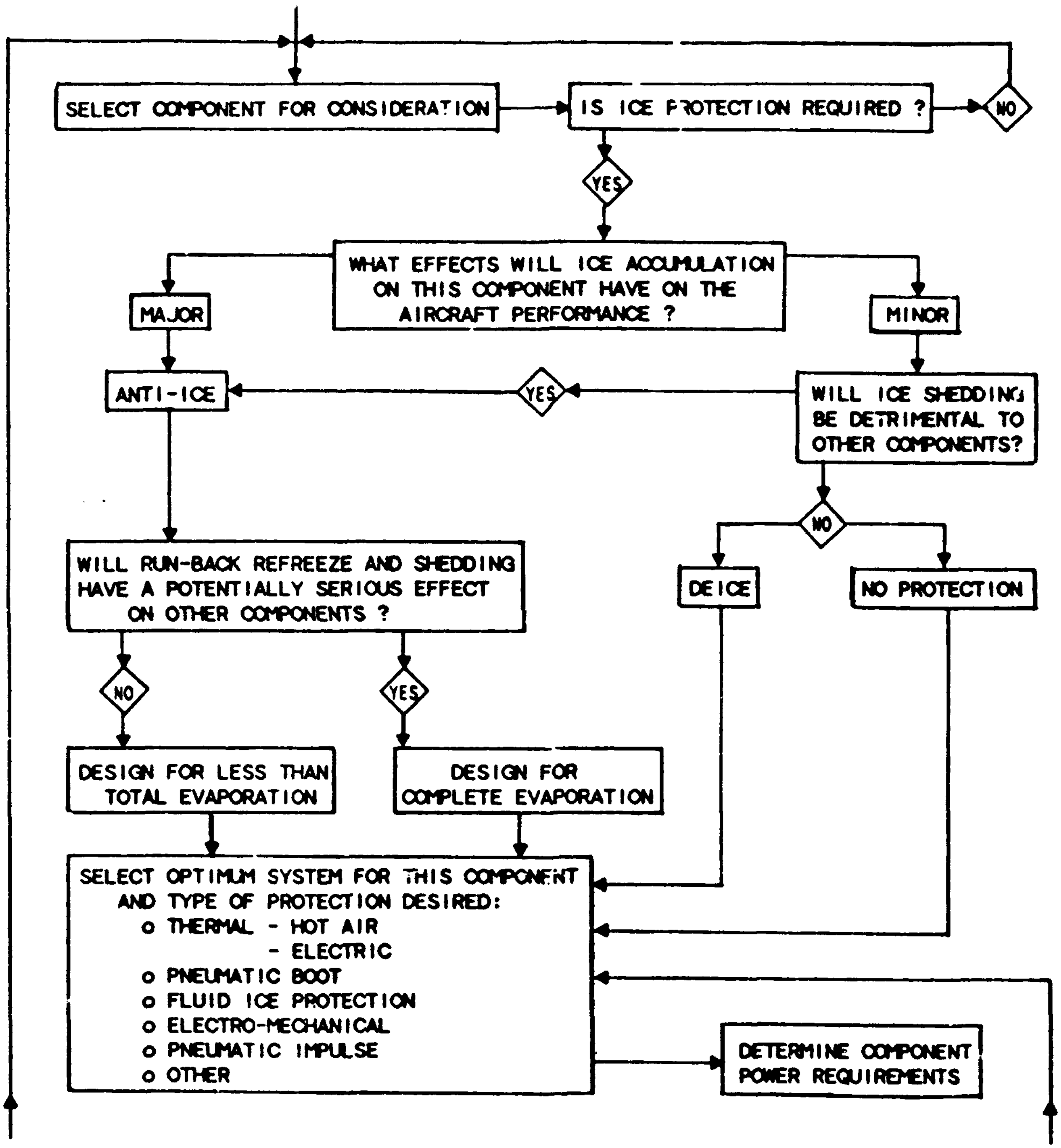

FIGURE 6-1. SYSTEM SELECTION FLOW DIAGRAM

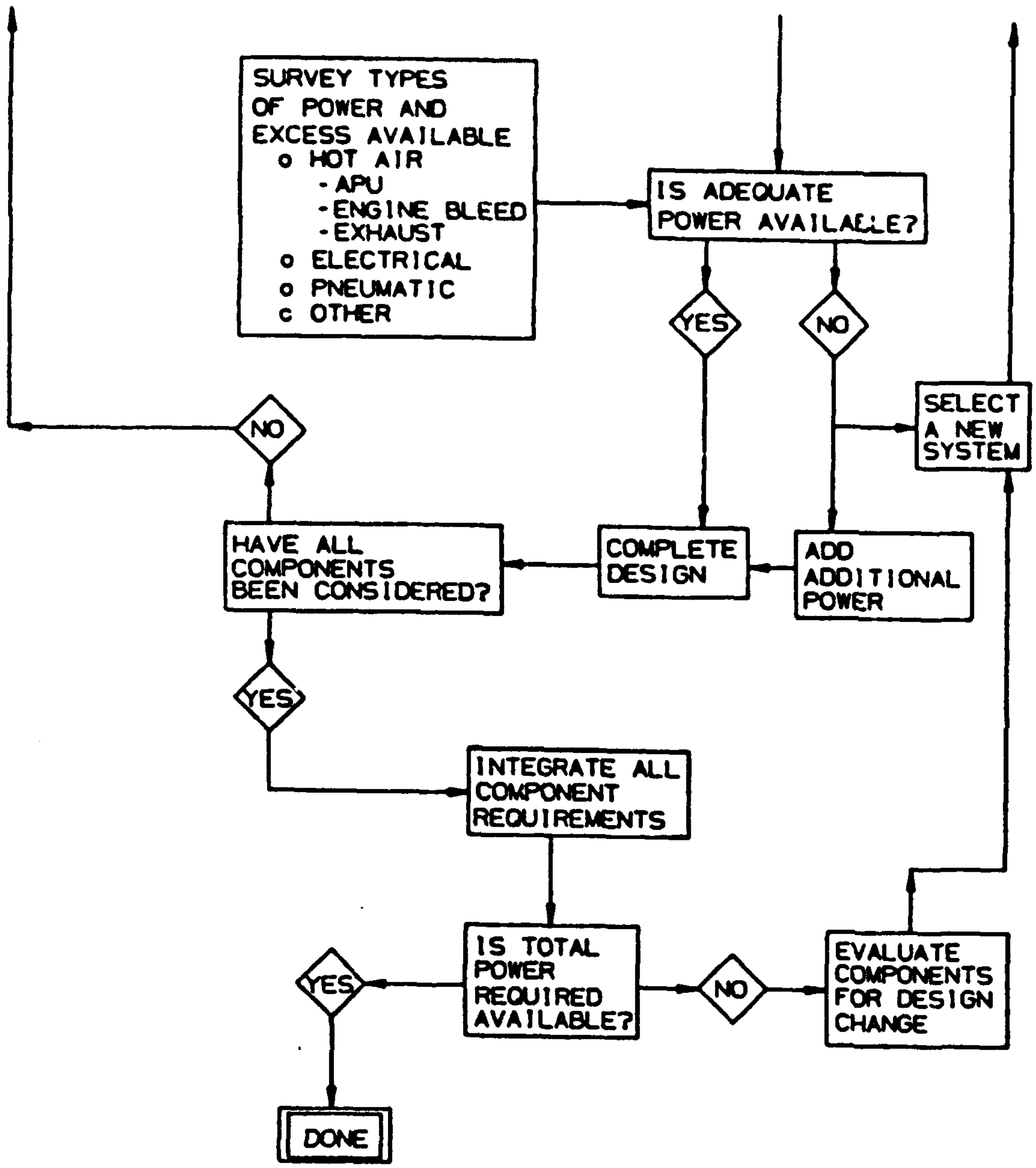

FIGURE 6-1. SYSTEM SELECTION FLOW DIAGRAM (CONTINUED)

CHAPTER IV - ICING SIMULATION METHODS

CONTENTS

SECTION 1.0 TEST METHODS AND FACILITIES

LIST OF TABLES

Note: These tables do not reflect any changes since 1987. However, some of the information in the text, in particular the discusssion of the NASA Lewis Icing Research Tunnel, is current as of December 1990.

These data are required as a function of flight time in the icing cloud, so time correlation of photographic data with the other data is essential. A data acquisition system is needed to collect and reduce the data for display in near-real-time. A typical data acquisition system would include a micro-processor, a data recording system, a printer or CRT for near-real-time output, and an operator control panel.

1.4.4.2 Factors Affecting Icing Simulation

Icing testing behind a tanker aircraft is difficult to control and often requires the coordination of three aircraft. The tanker aircraft lays down an icing cloud as the lead aircraft in the formation. A chase aircraft (the second aircraft) is often used to obtain photographs, visual observations, and to aid in positioning the test aircraft vertically in the icing cloud. The third aircraft is, of course, the test aircraft, which must be held in the icing cloud at the desired position.

The location of the test aircraft in the icing cloud is vital to the success of the test. Variations in the LWC and droplet size occur both in the vertical and axial positions within the cloud. This is illustrated in figures 1-15 and 1-16. Although these two figures are for the U.S. Army's Helicopter Icing Spray System (HISS) (reference 1-28), they are representative of icing clouds produced by tanker aircraft.

The variations in LWC and droplet size make it difficult to determine if the aircraft is being tested at the desired critical condition. For example, if the test condition is selected to have an LWC of 0.9 g/m^3, figure 1-15 indicates that a difference of about 2 feet (.6 meters) in altitude from that which is desired can result in LWC varying as much as 0.8 g/m^3 from the desired value. The same type of analysis can be applied to the variation in droplet size as shown in figure 1-16.

The temperature of the icing cloud is an important meteorological parameter in setting icing conditions. Unless the tanker tests are conducted during the winter months, the desired atmospheric temperatures must be achieved by flying at altitudes that are higher than normal. Under these conditions, both the pressure altitude and humidity generally do not agree with those of a naturally occurring icing cloud. To compensate for the variations, changes should be made in the LWC and droplet size of the simulated cloud. The magnitude of the change can be calculated by the procedure of reference 1-23. However, to do so requires a prior knowledge of the pressure altitude and humidity encountered during testing. Frequently this is unknown before testing so compensation for the effects of pressure altitude and humidity are not taken into account. For this case, testing behind a tanker becomes an uncontrolled experiment, and the data acquired may not represent the critical icing design condition as originally assumed.

Altitude versus airspeed envelopes are given for several tankers in figure 1-17.

IV.1.5 SIMULATED ICE SHAPES

A method of assessing the effect of ice on the performance and handling of an aircraft is flight test in dry air with ice shapes affixed to the aircraft. These simulated shapes have been made of wood, styrofoam with fiberglass covering, and a molded plastic. The surface roughness of ice may be simulated by adding body putty or epabond to the surface (reference 1-29).

The shapes can be found from icing tunnel tests, earlier icing flight tests, or an analytical model computation. For safety, some aircraft have had the ice models attached to their wings by quick-release fasteners.

IV.1/U1 TEST AIRCRAFT-MOUNTED AISS

An Airborne Icing Spray System, AISS, is a spray system which is mounted on the test aircraft, typically a helicopter. Bleed air and water must be available in sufficient quantities to feed the pre-calibrated spray nozzle arrays mounted approximately seven feet ahead of an engine inlet or other component. Due to the short dwell time before impact, droplet size is not believed to be appreciably affected by evaporation, a potential advantage over a tanker generated icing cloud. Due to the short distance the icing cloud must traverse and the use of a movable shroud around the nozzle arrays, icing cloud controllability may be potentially improved over that obtainable by a tanker aircraft setup. Icing cloud turbulence may be greater with the test aircraft mounted AISS because of the proximity of the spray nozzles themselves and the influence of the nozzle mounting system. Ice shapes observed during one test series suggested that icing cloud turbulence may not be a significant factor with respect to ice shapes.

This method has been used in the testing of an engine air induction system on a twin engine helicopter. In this particular application, droplet size was established using oil slides which were photographed within seconds of exposure. A shockmounted microscope equipped with a special slide holder was used for the droplet size determination. Liquid water content was also measured.

This test setup may be potentially useful for the development and certification testing of relatively small aircraft components such as engine inlets. As compared to other testing methods, the test aircraft mounted AISS may allow aircraft components to be qualified for icing conditions expeditiously as well as economically. Methodology, test setup and results are described in detail in reference 1-U1.

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Navier-Stokes solver - A Navier-Stokes solver is a computer code which implements a numerical method for the solution of the Navier-Stokes equations or a simplified version of those equations (e.g., the thin Navier-Stokes equations).

panel method - A panel method is a numerical method for calculating the incompressible, inviscid flow about a two- or three-dimensional body or configuration of arbitrary shape. The body is represented by panels (which may be line segments in two dimensions and planar regions bounded by polygons in three dimensions).

potential flow - Given a flow field specified by a vector function V. If V can be written as

$$\vec{V} = \nabla\phi$$

for some scalar function ϕ, then the flow is referred to as a potential flow (and ϕ is called the potential function). Invsicid, irrotational flows are potential flows.

Reynolds number - The Reynolds number is a dimensionless parameter which is the ratio of the inertia force to viscous force for a given problem. It is calculated according to the formula

$$Re = \frac{\rho_a V L}{\mu}$$

where V is a characteristic velocity and L is a characteristic length for the problem and the other symbols are defined in "Symbols and Abbreviations."

stagnation point - The point on a surface where the local free stream velocity is zero. It is also the point of maximum collection efficiency for a symmetric body at zero degrees angle of attack.

Thin (or thin-layer) Navier-Stokes equations - The thin (or thin-layer) Navier-Stokes equations are an approximation to the Navier-Stokes equations in which viscous terms containing derivatives in the directions parallel to the body surface are neglected.

Weber number - The Weber number is a dimensionless parameter which is the ratio of the inertia of air to the surface tension force at the air/water interface for a given problem. (This definition can be generalized to any two fluids.) It is calculated according to the formula

$$We = \frac{\rho_a V_\infty^2 L}{\sigma}$$

where V is a characteristic velocity and L is a characteristic length for the problem and the other symbols are defined in "Symbols and Abbreviations."

ß-curve - See "impingement efficiency curve."

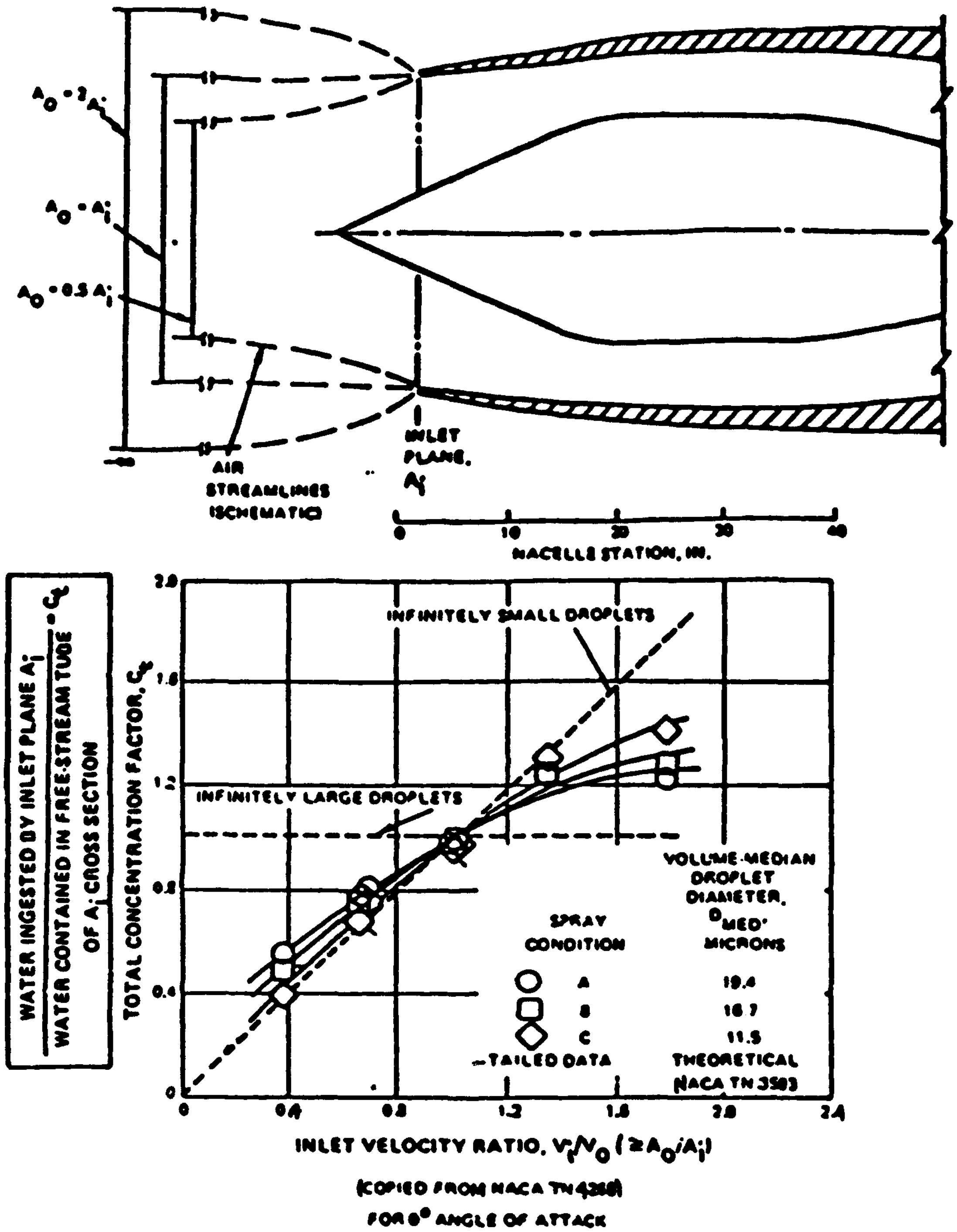


FIGURE 2-28. INLET CATCH EFFICIENCY (Reference 2-128)

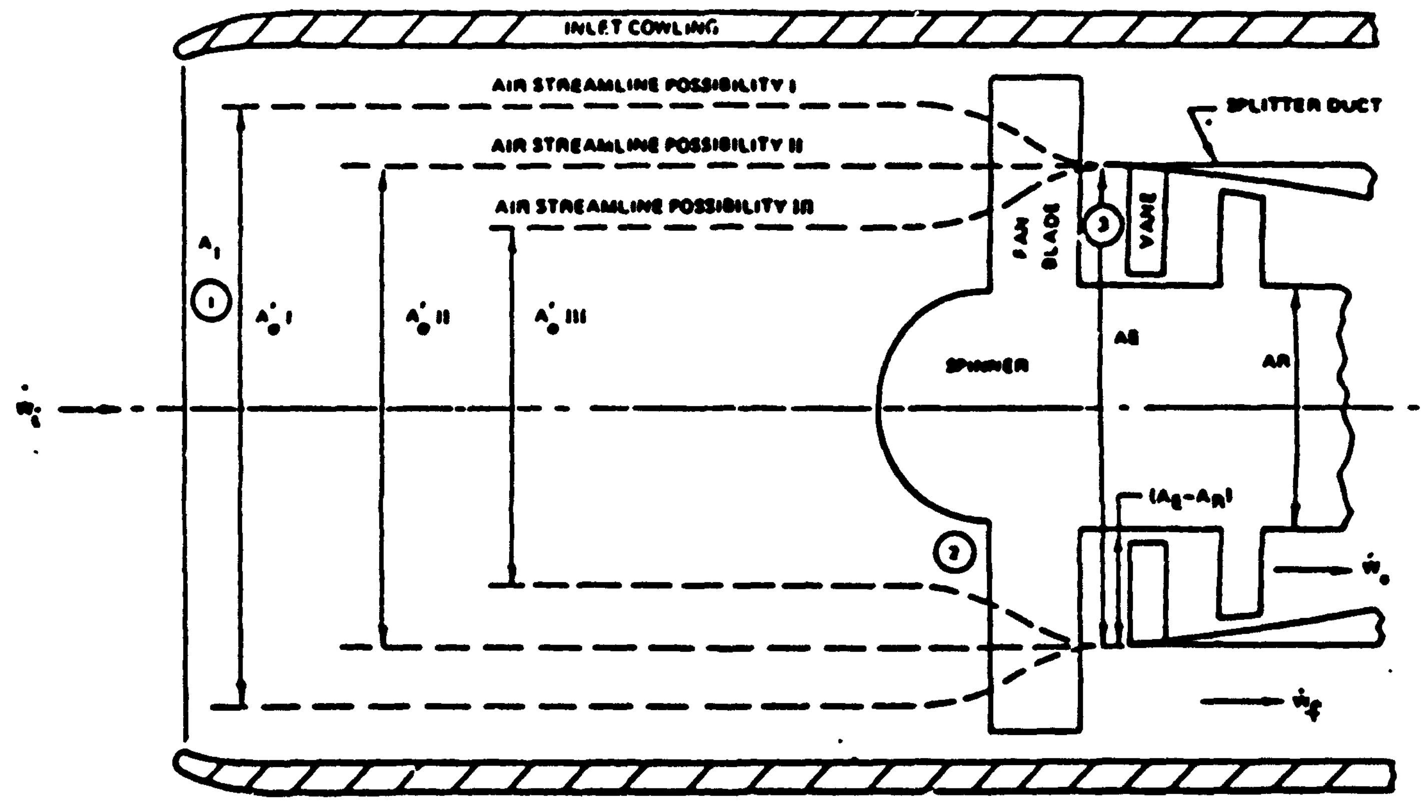


FIGURE 2-29. INLET STREAMLINE POSSIBILITIES FOR SPLITTER DUCT BEHIND A FAN (Reference 2-128)

CHAPTER V - DEMONSTRATING ADEQUACY OF DESIGN

CONTENTS

SECTION 4.0 TESTING TO DEMONSTRATE COMPLIANCE

LIST OF TABLES

TABLE 4-17. HELICOPTER INLET ICING TEST

Introduction

An Aircraft Icing Spray System (AISS) has been used to develop and certify a newly designed engine inlet for the Bell 222/250-C30G helicopter conversion. The air induction system was to comply with the specifications set forth in FAR 29.1093 and advisory circular AC-29-2A.

Icing Cloud Generating Equipment

The AISS used consisted of a spray rig, water and air supply, and an instrumentation package. It was capable of producing simulated supercooled icing clouds with MVDs ranging from 19 to 58 μm and LWCs from .58 to 2.6 g/m^3.

The spray rig was made up of two arrays of 9 and 25 internal mix nozzles. To achieve the required LWCs, these arrays were set up so that they could be operated separately or together. The spray nozzle tree holding these arrays was enclosed in an adjustable shroud, which was set to position the icing cloud with respect to the item to be tested. The entire package was mounted on the helicopter roof ahead of the starboard engine inlet. The icing cloud could be observed by the pilot using externally mounted mirrors.

The water supply consisted of a 30 gallon water tank feeding a double-acting, variable, positive displacement pump. Any ice breaking loose as the rotor craft enters non-icing conditions is contained by the coarse screen (figure B, 7).

Outside air temperature (OAT) was measured adjacent to the ship's OAT sensor, with aft facing thermo-couples mounted on the ice wand (figure B, 3) and foreign object damage (FOD) screen (figure B, 5). Temperature and pressure of the water and air supply were obtained at the entrance to the spray rake.

Inlet losses due to icing were estimated using engine turbine outlet temperature (TOT) rise, a single Pitot static tube mounted at the engine inlet and blockage calculations based on test results.

Oil slides and holders were designed such that droplet samples could be obtained through a roof mounted tube and photographed within three to four seconds of exposure.

Various valves, pump controls and a flow meter completed the instrumentation requirements.

Results

Testing of the Bell 222 helicopter employing the AISS was conducted between February 23 and March 15, 1989, in International Falls, Minnesota (cold temperatures) and on March 18 and 19, 1989 in Ames, Iowa (warm temperatures) The following is a typical flight profile:

1. Aircraft is fueled up to maximum gross weight.
2. After engine startup, bleed air is fed into air water supply lines to prevent system freeze-ups.
3. The estimated water flow rate required for the first part of the test is set during climb-out.
4. Aircraft climbs to the desired OAT level and then levels out at the test airspeed.
5. Bleed air pressure is then set to estimated value and water is directed into spray rig.
6. Bleed air to water line is disconnected.
7. Water flow rate is now adjusted, if required, to desired liquid water meter reading.
8. Using oil slides, ice cloud droplet samples are taken, checked for size and photographed using a shockmounted microscope. If necessary, bleed air pressure is adjusted and the procedure repeated.
9. All required aircraft and spray rig data are manually recorded. Videos of the inlet screen are taken through the aft cabin window.
10. Steps 8 through 10 are repeated for the second part of the conditions.
11. After all required data are taken, the water supply line and water passage in the spray rig are purged with bleed air.
12. After landing, the ice buildup on various induction system parts is observed and photographically documented.

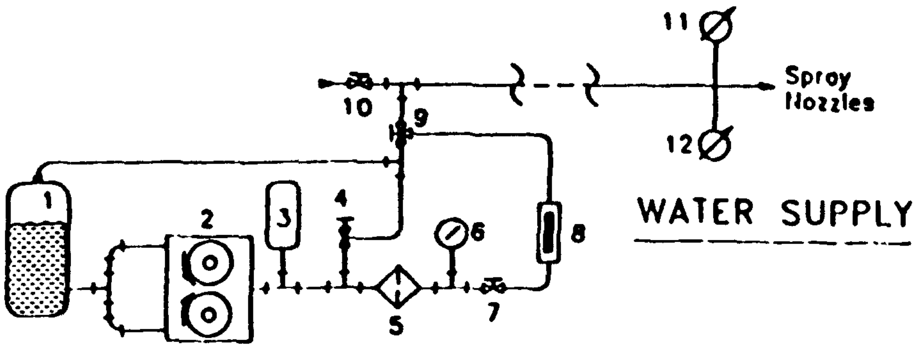
Test conditions flown are listed in table A. Droplet sizes produced by the AISS were observed and photographed using a shock mounted microscope and oil slides. Subsequent analysis has shown that droplet size targets were met at planned or higher LWC levels. Furthermore, it was found that the size distribution of the "small" droplet runs compare fairly well with that encountered in a "standard" stratiform cloud.

The ice buildup on the inlet screen was photographed using a video camera via a wing-mounted mirror. Still pictures were taken on the ground after each run to record final ice configuration on inlet and internal surfaces.

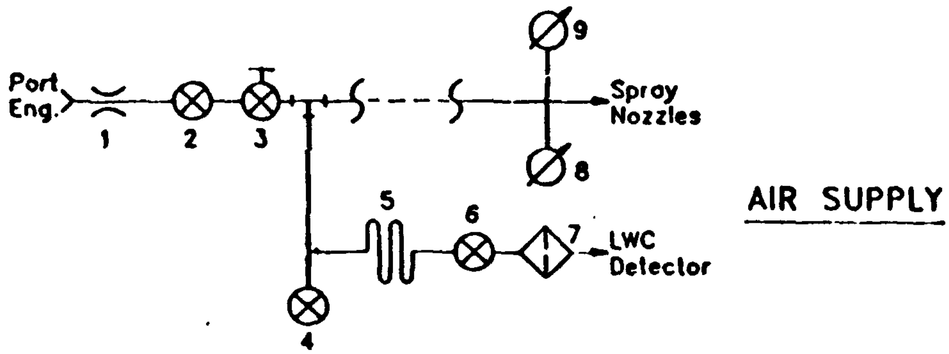
The FAA concurred, based on the observed test results, that the air induction system flown will adequately protect the engines from detrimental ice buildup during an inadvertent encounter of icing conditions.

Table A. Actual Test Conditions Flown

COND.	KIAS	OAT	TIME	LWC	MVD
5C	75	16.2	30.0	1.02	21
9B	50	31	22.0	1.77	42
5A	50	16.6	5.3	2.62	19
			21.7	1.28	22
5B	100	14.7	6.4	2.13	28
			12.4	0.71	25
5E	50	16.7	5.3	1.34	58
			21.9		
5D	100	18.7	6.4	0.80	40
			12.4	0.58	41
8A		23.6	40.0	0.88	43
9A		32.0	30.0	2.10	
7A	50	−4	5.3	2.46	19
			21.9	1.08	17
7B	100	−2	6.4	1.74	25
			12.4	0.71	25
6A	50	10.5	27.0	1.94	26
6B	50	14	21.9	1.12	25
			5.3	1.86	25
6D	50	15	21.9	0.90	29
			5.3	2.15	21

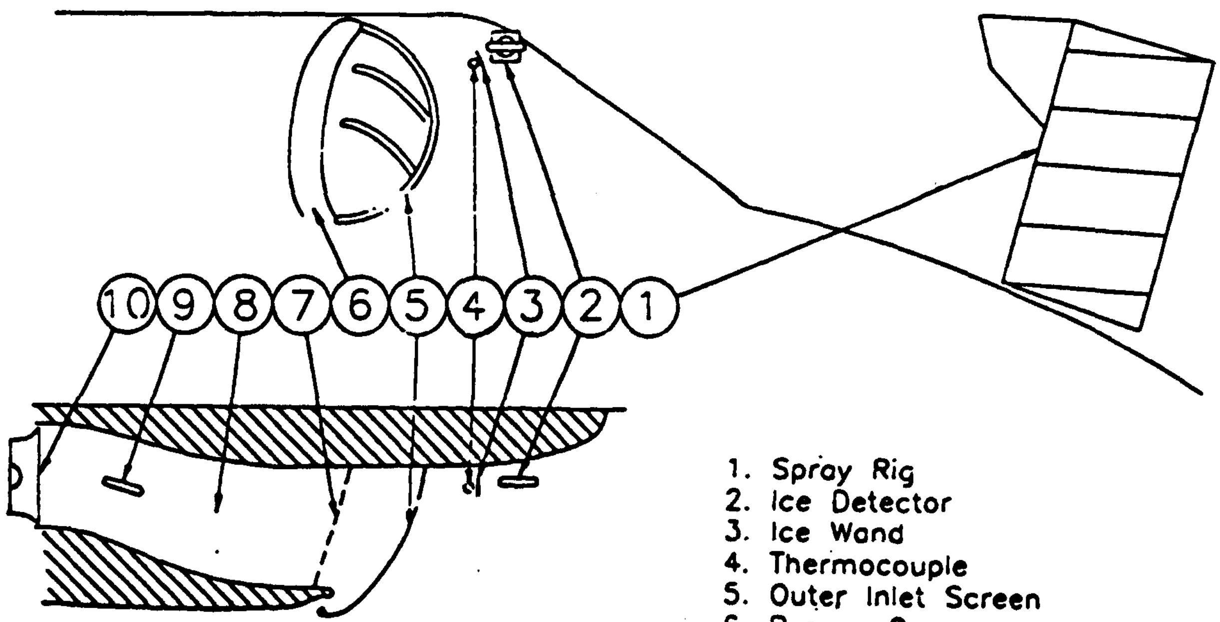


1. 30 gallon ventilated water tank
2. Double-acting, variable, positive displacement water pump
3. Accumulator
4. Pressure relief valve
5. Water filter
6. Pressure guage
7. Metering valve
8. Flow meter
9. Three-way valve
10. Bleed-air shut-off valve
11. Thermocouple
12. Pressure pick-up



1. Bleed air orifice (d = 0.435 in.)
2. Bleed air shut-off
3. Bleed air control valve
4. Water trap
5. Cooling coils
6. Shut off valve
7. Filter for LWC meter (see inst. man.)
8. Thermocouple
9. Pressure pick-up

Figure A. Test Equipment Schematic



1. Spray Rig
2. Ice Detector
3. Ice Wand
4. Thermocouple
5. Outer Inlet Screen
6. Bypass Gap
7. Inner Screen
8. Air Duct
9. Pitot Tube
10. C30G Engine Inlet

Figure B. Test Aircraft Configuration and Setup (Schematic)

The icing envelopes for Transport Category helicopters is contained in FAR Part 29 Appendix C. These are identical to those of FAR Part 25 Appendix and are presented in figures 1-1 through 1-6. These icing envelopes have served as a satisfactory design criteria for fixed wing operations in icing conditions for over two decades. The envelopes extend to 22,000 feet (6700 m) with possible extensions to 30,000 feet (9140 m) and does not present icing severity as a function of altitude. At the time the envelopes were derived, it was assumed that all transport category airplanes would operate to at least 22,000 feet. For present state of the art rotorcraft, this assumption is not valid. Thus an altitude limited icing envelope, based on the same data used to derive the FAR part 25 (reference 1-2) Appendix C envelope, is presented in FAA AC 29-2 as an alternate to the full icing envelope. These envelopes are reproduced and presented as figures 1-9 through 1-12. In addition, recent work by Masters (reference 1-13) recommends a new characterization for altitudes below 10,000 feet (3048 m). These envelopes, as compared to FAR Part 25 Appendix C, are presented in figures 1-7 through 1-8. Neither of these latter envelopes are regulatory but are offered as options.

1.2.5 FAR Part 33 (Engines)

In order to become certified for flight by the Federal Aviation Administration, aircraft engines must demonstrate (by test, analysis or similarity) that they are capable of operating successfully in icing conditions (reference 1-5). In addition, the gas turbine engine must be capable of withstanding the foreign object ingestion test of FAR 33.77 without failure or hazard. The engine should be designed and demonstrated to be capable of ingestion of the most severe ice accumulation that could occur for the particular installation. The pertinent paragraphs of FAR Part 33 are listed in Section 1.1.

FAR paragraph 33.66 specifies that if bleed air from the engine is used for engine anti-icing and can be controlled, provision must be made for a means to indicate to the flight crew that the engine ice protection system is operating.

FAR 33.68 specifies the icing requirements for engine induction systems. The FAR Part 25 appendix C icing conditions apply and were discussed in Chapter I and are presented in figures 1-1 through 1-6 (reference 1-1). FAR 33.68(a) requires that the engine must operate throughout its flight power range without the accumulation of ice on engine components that would adversely affect engine operation or that would cause a serious loss of power or thrust in continuous maximum and intermittent maximum icing conditions. FAR 33.68(b) requires that the engine must be able to idle for 30 minutes on the ground with available air bleed for ice protection at its critical condition without adverse effect in a specified atmospheric condition, followed by a momentary operation at takeoff power or thrust.

The non-specific nature of FAR 33.68(a) allows each engine manufacturer to work out a set of mutually agreeable compliance tests with the Federal Aviation Administration pertaining to his specific engine in his specific test facility (reference 1-21).

FAR 33.77 states the foreign object ingestion requirements for engines. Paragraph 33.77(c) states that ingestion of water, ice, or hail, under prescribed conditions, may not cause a sustained loss of power or thrust or require the engine to be shut down. The ice ingestion requirement is presented in table 1-1. Paragraph 33.77(d) states that if the engine incorporates a protection device (e g , a screen) in the engine inlet, then the ingestion requirement is waived if the ice cannot pass through the protective device, the protective device will withstand the impact of the ice, and if the ice stopped by the protective device does not obstruct the flow of induction air into the engine with a resultant loss of power or thrust greater than those values specified in FAR paragraph 33.77. Experience has shown that ice can build up on the back side of such screens with the potential of engine damage should the ice shed from the screen.

In general, ice protection systems on engines intended for installation in helicopters are subject to the same standards as for fixed wing aircraft engines. Some interesting helicopter icing phenomena which apply in a secondary manner to the engine are reported in reference 1-21 (FAA-RD-77-76, page 6-7).

VI.1.3 REFERENCES

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TABLE 1-1. ICE INGESTION REQUIREMENTS FOR TURBINE ENGINES (REFEFENCE 1-5)

Foreign Object	Test Quantity	Speed of Foreign Object	Engine Operation	Ingestion
Ice	Maximum accumulation on a typical inlet cowl and engine face resulting from a 2-minute delay in actuating anti-icing system, or a slab of ice which is comparable in weight or thickness for that size engine.	Sucked in	Maximum cruise	To simulate a continuous maximum icing encounter at 25°F.

NOTE: The term "inlet area" as used in this section means the engine inlet projected area at the front face of the engine. It includes the projected area of any spinner or bullet nose that is provided.

CHAPTER VIII

BIBLIOGRAPHY

VIII.1.0 INTRODUCTION

This Aircraft Icing Bibliography was originally prepared using the bibliography created by Dr. Ken Korkan of Texas A & M University for the Society of Automotive Engineers (SAE) AC-9C Subcommittee (SAE ATR-4015) on Aircraft Icing Technology. Additional references have been incorporated to make the bibliography more complete concerning aircraft icing.

The principal sources for the original bibliography were as follows:

(a) Bibliography of Unclassified National Research Council of Canada Aircraft Icing Reports and Publications.

(b) K. D. Korkan, "Compendium of Aircraft Anti-ice/Deice/Ice References," private communication, Texas A & M University, College Station, Texas, 1983.

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(f) "Ice Protection Investigation for Advanced Rotory Wing Aircraft," Bibliography prepared under contract DAAJ02-72-C-0054, USAAMRDL Technical Report.

(g) As provided by the members of the Aircraft Icing Technology Subcommittee.

(h) Defense Documentation Center (DDC).

(i) National Technical Information Center (NTIS).

The Aircraft Icing Bibliography has now been converted to the Aircraft Icing References Database, ICE_RFS, in RBASE. The current bibliography published here was generated from ICE_RFS using an RBASE program. Approximately 300 titles have been added. The format of the sections has been slightly modified. All entries in a section dated 1959 or later are now printed first as Part A; all entries dated 1958 or earlier, or not dated, are printed second as Part B. 1959 was the year that the NASA succeeded the NACA.

An "ACSII image" (ICE_RFS.ASC) of ICE_RFS.ASC has been generated using the RBASE utility Gateway. ICE_RFS.ASC can be imported by <u>any</u> database program. Thus the icing references database is available to anyone with a database program who requests the file. The structure is very simple and is described on the next page:

STRUCTURE OF AIRCRAFT ICING REFERENCES DATABASE
ICE_RFS.RBF (RBASE)
ICE_RFS.ASC (ASCII)

FIELD	WIDTH(MAXIMUM)	
AUTHORS	110	Enough for 7 authors unless they have unusually long last names or a lot of leading initials.
TITLE	200	Long enough for any title currently included.
INFO	250	Information such as journal name, volume, number, and date for an article, or institution, number, and date if a report.
YEAR	4	This field is actually redundant, since year should appear in the INFO field. However, it should be useful for database searches.
SECTIONS	15	The bibliography sections (topics) to which the entry pertains.
REVISION	3	This field is for keeping track of updates.

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U.S. GOVERNMENT PRINTING OFFICE: 1994 - 504-078/80231